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FOREWORD

This document constitutes a portion of the final report (SID 65-500) for Addendum 1 of Contract NAS9-3140, Extended Apollo Systems Utilization Study, dated 6 July 1964, prepared by the Space and Information Systems Division of North American Aviation, Inc. The analyses described herein and in the volums listed below were conducted under the direction of the National Aeronautics and Space Administration's Manned Spacecraft Center as an addendum to the basic contract which included the Apollo X Study (SID 64-1860) and the Prolonged Missions Study (results to be published later). This final report has been prepared in a series of five volumes as follows:

1. Summary

2. Experiment Analysis and Requirements

   Part I: NASA Flights
   Part II: Air Force Flights
   Appendix A. NASA Experiments
   Appendix B. Air Force Experiments
   Appendix C. Mission Scheduling Computer Printout for NASA Flights

3. Configuration Analysis and Experiment Accommodation

   Appendix A. Engineering Drawings
   Appendix B. Air Force Missions

4. Subsystems Analysis

5. Development Planning

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APOLLO EXTENSION SYSTEMS, SPACECRAFT/EXPERIMENT INTEGRATION, BLOCK II APOLLO, BLOCK IIA APOLLO, APOLLO X, APOLLO MODIFICATIONS, CONFIGURATION C, CONFIGURATION D, CONFIGURATION 1, CONFIGURATION D', RACK, PALLET, AUXILIARY MODULE

ABSTRACT

This report defines the design characteristics of four potential extended mission Apollo spacecraft designed to specific NASA mission and configuration requirements. All configurations are for Earth-orbital missions, which include low inclination, polar, and synchronous orbits and comprise a total of 15 NASA missions.

The four configurations studied present various degrees of modification to the Apollo Block II CSM. Configuration 1 is employed for 14-day missions and requires minimum modification. It may be used with either an experiments rack or pallet or both. Configuration C is identical to the Apollo X CSM and is utilized with an experiments rack for durations of up to 45 days. Configuration D is essentially a Block II CSM, with subsystems for life extension installed in the experiments rack. Additionally, an experiments pallet may be installed in the Configuration D CSM. Configuration D' is for 30-day missions and uses the Block II CSM as a baseline, similar to Configuration D. Early schedule requirements, however, dictate the prototype modification of critical subsystems in Configuration D' in order to meet the 30-day mission requirement.

The capability of each configuration was measured against the requirements dictated by 15 NASA-derived experimental packages. Additionally, the Air Force MOL experiments were examined and optimally grouped in a minimum number of flights. It was found that all NASA missions could be accommodated within the capabilities of the applicable configurations with certain revisions to experiment operation and/or mission duration to meet launch vehicle payload limits. All Air Force experiments could be accommodated in five flights.

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PREFACE

Over the past two years, a number of investigations have been performed to determine the characteristics of modifications required to extend the orbital duration of the Apollo spacecraft for alternate mission applications. Initial studies examined system characteristics for application as a 120-day Earth-orbital laboratory vehicle. Because of this extended duration, it was necessary that advanced subsystem concepts be employed in several cases to remain within the payload capabilities of the Saturn IB launch vehicle. Subsequent studies determined the characteristics of the Apollo CSM assuming restriction to use of only current Apollo subsystems concepts. Under this restriction, it was found that the Earth-orbital duration capability of the CSM was limited to approximately 90 days because of Saturn IB payload limits.

The recently concluded Apollo X study examined in depth the CSM characteristics and modifications required to perform NASA near-term missions of interest. These missions included both extended Earth-orbital and lunar-orbital mission durations of 45 and 34 days, respectively. Concurrent with the Apollo X study, separate contractors were examining the characteristics of: (1) the LEM ascent stage modified for use as an experiment module, and (2) an experimental laboratory module of new design.

Since the conclusion of the Apollo X study, several configuration innovations have been conceived by NASA which could warrant inclusion in the Apollo Extension Systems (AES) program. Two of these innovations take the form of experimental modular appendages and are identified as a "rack" and a "pallet"; the rack could be used in place of the modified LEM or a new laboratory module, and the pallet could be installed in the empty bay (Sector I) of the service module. Using these modular appendages, the alternate mission capability of the Block II Apollo could be increased through the addition of experiments as well as subsystems required for mission life extension beyond 14 days. It became apparent that further studies were warranted relative to (1) the comparative operational effectiveness of the various experimental appendages, and (2) the possible methods for extending the orbital duration capability of the CSM. The method selected to achieve orbital-life extension could actually result in varying degrees of CSM changes—depending upon the subsystem extension philosophy implemented.

A matrix of configurational approaches could readily be defined as only partially indicated in the accompanying chart, AES Concepts (Figure 1), with corresponding variations in costs, schedules, and operational capabilities. Therefore, in order to evaluate the characteristics and capabilities of each of the

CSM CONFIGURATIONS	EXPERIMENTAL APPENDAGES		
	S&ID	GRUMMAN LEM A/S	BOEING LAB MODULE
I BLOCK II	RACK EXPERIMENTS ONLY; EXPERIMENT PALLET	EXPERIMENTS ONLY	EXPERIMENTS ONLY
C APOLLO-X	RACK EXPERIMENTS ONLY	EXPERIMENTS ONLY	EXPERIMENTS ONLY
D BLOCK II MODIFIED	RACK EXPERIMENT AND SUBSYSTEMS; EXPERIMENT PALLET	EXPERIMENTS AND SUBSYSTEMS	EXPERIMENTS AND SUBSYSTEMS

Figure 1. AES Concepts

possible combinations, parallel AES studies were initiated by NASA. S&ID was responsible for evaluating the characteristics of the CSM, rack, and pallet; Grumman and Boeing conducted separate studies of the LEM experimental module and new laboratory module design, respectively. The results of the three contractor studies were to be in a form such that the design and development characteristics of all possible system elements could be assembled by NASA into complete configurations and development programs of their own choosing.

The S&ID study was concerned with the examination of several basic configuration approaches, each of which represents varying degrees of Block II CSM modification, experimental capability, operational complexity, and program costs. These configuration approaches are defined as follows:

CONFIGURATION 1

By NASA definition, Configuration 1 is essentially the Block II CSM—without major changes, but with the addition of an experimental pallet (in SM Sector 1) and/or an experimental appendage (rack or laboratory module) docked to the CM during orbital operations. Orbital life is necessarily limited to 14 days or less since changes to the CSM subsystems are precluded. Subsystems support for the experiments—except for those in the pallet—is provided by the CSM.

CONFIGURATION C

Configuration C is identical to the CSM approach derived in earlier Apollo X studies where mission life extension to 45 days was achieved through

the addition (in the CSM) of expendables, spares, and redundancies of Apollo Block II subsystems. Since the service module Sector 1 is occupied by subsystems in this approach, the pallet does not apply here. However, the rack or either of the laboratory modules would be included. As in Configuration 1, support of the experiments housed in the rack or laboratory module is provided by the subsystems located in the CSM.

CONFIGURATION D

Configuration D was to be based upon use of the Block II CSM with minimum modifications, in combination with an experiments/subsystems rack or laboratory module and with a pallet if required. Experiment support is provided by the CSM subsystems during the first 14 days (approximately) of orbital operation, after which subsystems life extension provisions installed on the (experimental appendages) would provide support both to the CSM and the experiments for the remainder of the 45-day orbital duration. By definition the subsystems installed on or in the experimental appendage were to be of the type defined under the prior Apollo X study; these included, for example, extended life fuel cells with in-space start, new cryogenic tankage, etc. During the early phases of the study, the Configuration D approach was modified through mutual agreement by NASA and S&ID at a series of weekly interface meetings. As a result, additional Configuration D ground rules were —of necessity—formulated which yielded a configuration that does not absolutely adhere to the requirement that the Block II CSM remain unchanged. More explicitly, revised Configuration D ground rules resulted in the installation of only the power system life-extension capability on the external device.

CONFIGURATION D'

Additionally, a requirement for one early 30-day mission (Flight 211), dictated the establishment of a unique approach—which was identified as Configuration D' by S&ID. This configuration is similar to Configuration D with respect to subsystem location and general arrangement. However, in Configuration D' the mission life-extension is provided by the use of only Block II subsystems that are "stretched" through prototype modifications to accomplish the required 30-day mission. The cryogenic storage system included on the rack, for example, is comprised of multiples of Block II cryogenic tanks rather than of the new and larger tanks defined in the Apollo X study.

A summary of characteristics of the four configurations of interest is presented in Table 1. A detailed definition of the vehicle and subsystems ground rules for each configuration may be found in appropriate volumes of this report.

Table 1. Configuration Characteristics

		Configuration 1 (14 Days)	Configuration D' (30 Days)	Configuration C (45 Days)	Configuration D (45 Days)
Command Module	Structure	Block II	Block II	Block IIA	Block IIA
	Subsystems	Block II	Block II modified	Apollo X	Apollo X
	LiOH and crew systems	1 day	1 day	1 day	1 day
	Umbilical to rack				
Service Module	Structure	Block II	Block II	Block IIA	Block II modified
	Cryogenics	Block II	Block II	Apollo X	Block II
	Fuel cells	Block II	Block II	1000 hours	Block II
	RCS	Block II	Block II	As needed	As needed
	SPS tankage	Block II	Block II	As needed	Block II
	Sector I	Empty or pallet	LMS installation	EPS installation	Empty or pallet
Rack (external device)	Cryogenics	None	Block II	None	Apollo X
	Fuel cells	None	Block II modified	None	1000 hours
	LiOH and crew systems	13 days	29 days	44 days	44 days

The design approach for each configuration was based upon consideration of the application of a standard vehicle capable of performing the defined NASA missions and Air Force experiments. Additionally, CSM/external appendage interfaces were mutually defined among NASA, S&ID, Boeing, and Grumman such that the CSM for each appropriate configuration could be used alternately with either the rack, pallet, LEM laboratory or separate laboratory module without change. S&ID studies of the rack were also oriented toward providing a common rack design for all configurations and missions which would be capable of containing experiments, or experiments and subsystems as required.

The primary objective of the study was to define the characteristics and capabilities of various CSM/rack/pallet combinations as applied to experimental flight package and mission constraints defined by NASA. The actual grouping of each of 15 experimental flight packages was the responsibility of IBM under the direction of NASA. Consequently, S&ID's role in the NASA experiments area was limited to defining the configuration and subsystems requirements demanded by each of these experimental flight groupings as a basis for the experiment integration task. The Air Force individually specified experiments were integrated into similar, but separate, vehicles; however, the optimal grouping of these experiments—in order to minimize the number of flights required—was accomplished by S&ID.

In order to assess the cost and schedule ramifications attendant upon each of the matrix considerations, development planning studies were performed based upon the NASA defined launch schedule (AE 65-1), which entails a total of 28 manned AES flights through the first quarter of 1972.

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INTRODUCTION

This volume presents the analyses and results of the subsystems investigations performed during the AES study. The subsystem definitions used are based upon Apollo Block II and life extension-associated revisions as determined in the previous Apollo X study. Of primary emphasis in the program were the ramifications attendant with NASA-defined ground rules relating to the location and operation of selected subsystems; i.e., relative to placement within the command-service module (CSM) or on the experimental appendage (rack, LEM, or laboratory module).

The initial philosophy established by NASA at the beginning of this study can be summarized for each spacecraft/subsystems configuration as follows:

Configuration 1 - Use Block II subsystems without change and minimize command module changes caused by interfaces with an external device. Because the Block II subsystems are designed for a mission of 14 days, Configuration 1 applies only to missions with nominal durations of 14 days.

Configuration C - Use Apollo X type subsystems (per earlier NAS9-3140 studies) in the CSM for life extension, and add only subsystems peculiar to experiments in the external device. Configuration C should have a nominal mission duration of 45 days.

Configuration D - Use Block II subsystems without change in the CSM, and use Apollo X type subsystems in the external device to both achieve life extension and to provide for experiment functions. This configuration should have a nominal mission duration of 45 days.

In addition to Configurations 1, C, and D special consideration was required for Flight 211 that calls for a 30-day duration scheduled early in the AES program. Because Configuration 1 cannot endure for 30 days and because Apollo X type subsystems would not likely be available for the scheduled launch of this flight, another configuration (D') was defined. Configuration D' uses Block II subsystems in the CSM and achieves the extended duration by placing additional Block II subsystems in the external device with prototype modifications of critical CSM subsystems.

Additionally, several other ground rules were postulated by NASA. First, a pure oxygen (5-psia) atmosphere was to be used in all configurations, and, second, the guidance and navigation system should be retained in all configurations on all flights. Thirdly, 1000-hour fuel cell stacks should be used in the external device for Configuration D and in the service module for Configuration C. Fourth, the external device should have an independent thermal control system for equipment and experiments housed therein.

The recommended Block II subsystem changes resulting from the previous Apollo X study are summarized in Table 2. Only those changes that could be substantiated by parametric analyses are presented. In other cases, certain changes are presently not indicated; however subsystem suitability must be verified through testing under Apollo X mission-simulated environment and duration. Therefore, Table 2 provides only an initial tabulation of required changes; testing may reveal the necessity for additional modifications.

Several comments regarding this modification list are apropos for clarification purposes. The Earth-Landing system requires the addition of volatile material to the parachute compartment to maintain the compartment pressure after 14 days. Therefore, this is a required change for Configuration D, D', and C. Because of the single-gas ground rule, the N_2 system (used in Apollo X studies) is not used in Configurations D and C.

The G&N and SCS modifications were either the result of extended durations or, as in the case of the lunar polar orbit mission, the result of severe mapping mission requirements. Some flights, particularly Flight 518, required a tighter attitude hold than was encountered in the Apollo X study. The Apollo X power system used 400-hour cells rather than 1000-hour cells; therefore, the number of fuel cell stacks required for a given mission success probability for the 45-day mission duration can be reduced.

As the study progressed, it became apparent that the initial NASA subsystems philosophy required modification. In Configuration D, the use of the external device as a means for subsystem extension proved meaningless or undesirable in some cases. For example, if the volatile substance for pressure maintenance of the parachute compartment were placed in the external device (with some sort of piping) an obviously undesirable complexity would result. A similar situation would exist with the addition of a redundant compressor (in the external device) for the CSM environmental control system unit and the resulting installation of piping between the two modules. Hence, additional ground rules regarding the Configuration D subsystem modifications were devised early in the study.

Table 2. Apollo X Subsystem Changes

Subsystem	Earth Orbit		Lunar Orbit	
	Low Inclination (45 days)	Polar (45 days)	Low Inclination (20-Day Total)	Polar (34-Day Total)
Comm/Data	Delete hi-gain antenna and rendezvous equipment.	Delete hi-gain antenna and rendezvous equipment.	No change	Delete rendezvous equipment.
ELS	Add volatile material.	Add volatile material.	Add volatile material.	Add volatile material and heaters
ECS	Modify compressor. Add cabin fans. Add N_2 system.	Modify compressor. Add cabin fans. Add N_2 system.	No change	Modify compressor. Add cabin fans and N_2 system.
G & N	Remove system	Use modified IMU and AGC	Use modified IMU and AGC	Use modified IMU and AGC
Power	5 Fuel cells with in-space start. New cryogenic tanks	5 Fuel cells with in-space start. New cryogenic tanks	Add 1 set Block II cryogenic tanks	5 Fuel cells with in-space start. New cryogenic tanks
Propulsion	Use small SPS tanks. One helium bottle only.	Use 1/2-size SPS tanks. One helium bottle only.	No change	No change
RCS	No change	Use LEM tanks (2 sets/quad).	No change	Use LEM tanks (2 sets/quad).
SCS	Modify electronics for G & N capability.	Modify electronics for horizon-scan system and redundancy.	No change	Modify electronics for horizon-scan system and redundancy.

The following additional ground rules have been utilized in the AES study:

1. Environmental control system (ECS)

Additional suit loop compressors will be installed in the CM. Suit system requirements will be considered, but no suit connections are required in the external device.

All LiOH and food, except for 3 man-days, will be stored in the external device.

Spare circulating fans will be stored in the CM.

All H_2 and O_2 in excess of 14 days are located in the external device.

Laboratory and LEM contractors will provide CM/external device atmosphere exchange.

No provisions shall be made for atmosphere exchange when the CM/external device interlock hatch is closed. It should be possible to maintain pressure for either module.

Extravehicular activities will be accomplished using the external device as an air lock. The LEM and laboratory contractors will provide gaseous oxygen as required for short-time repressurization.

The external device will provide for recharging the PLSS.

2. Stabilization and control system (SCS)

The G&N system will be incorporated for all flights whether required or not.

All subsystem changes and additions will be made in the CSM, including the storage of spares.

Coarse alignment of the vehicle for the experiments will be accommodated by the G&N. If required, the experiment will provide for fine alignment.

3. Reaction control system (RCS)

The service module RCS will be utilized as much as possible; service module tankage changes will be allowed.

The LEM, RCS, and SCS electronics will be utilized in conjunction with the CSM SCS (also for Configuration C).

Based on the preceding ground rules and philosophy, the subsystems studies established the required changes to the CSM subsystems and generated parametric data concerning the subsystems required on the external device (rack, LEM, or laboratory module).

COMMUNICATIONS/DATA SYSTEM

The differences between NASA and USAF ground station characteristics and experimental requirements require the communications and data subsystem to utilize two different types of equipment: one derived from the Apollo for NASA missions and one derived from the space-ground link system (SGLS) for USAF missions.

For NASA missions, the basic philosophy is to use Apollo communications and data subsystem in the command module and additional Apollo-type equipment in the external device to process the experimental data. Consequently, the NASA configuration is divided into three parts, namely the basic Apollo equipment, the standard equipment used for all NASA experiments, and the special equipment used for individual missions. This configuration is shown in Figure 2.

The special equipment comprises an experiment programmer and sequencer that receives ground commands from the standard equipment and routes them to the different experiments. This equipment also includes a signal conditioner that receives the data from the experiments and routes them to the standard equipment in the required form. Both experiments programmer and signal conditioner will be specially designed to be compatible for all experiments.

The standard equipment is basically existing Apollo equipment and is comprised of a command decoder, a PCM telemetry, a data recorder, an audio center, and a signal conditioner for Configurations D and D' only. A rendezvous radar also is included for Flights 219, 229, and 230 only.

The basic Apollo equipment is comprised of data, radio, and antenna equipments. The data equipment includes the signal conditioner, PCM telemetry, premodulation processor, television camera, central timer, data recorder, command decoder, and displays. The radio equipment includes the HF transceiver, VHF beacon, VHF-AM transceiver, S-band transceiver, and an S-band power amplifier. Both telemetry and tracking are handled by the S-band transceiver, exclusively. The antenna equipment includes a VHF multiplexer, S-band multiplexer, HF whip, VHF whip, VHF omnidirectional antennas, S-band omnidirectional antennas, and, for synchronous missions only, an S-band high-gain antenna. Both VHF and S-band omnidirectional antennas will be flush-mounted if possible. The high-gain antenna will be automatically oriented in the direction of the Earth by Earth sensors.

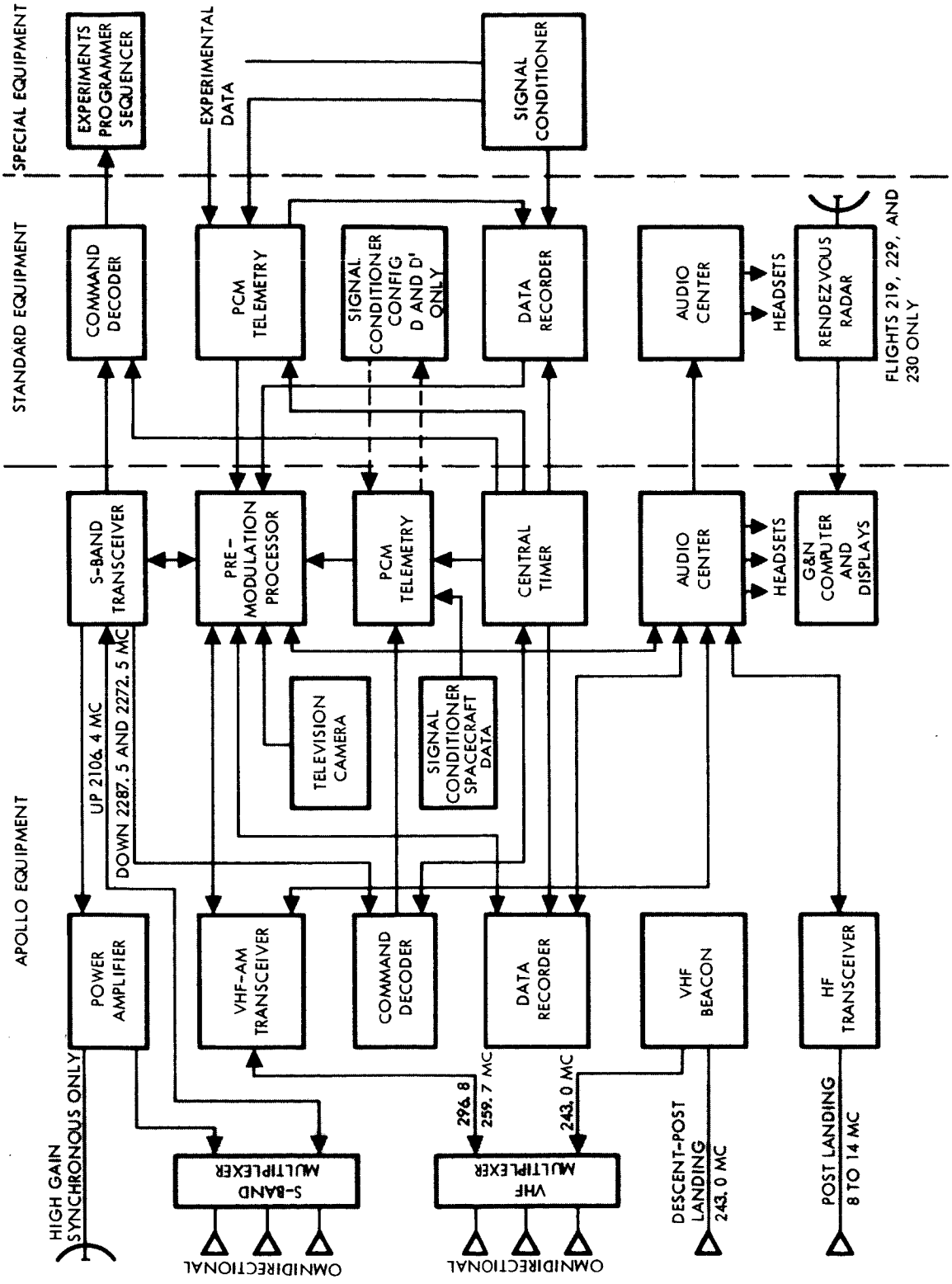

Figure 2. Communications and Data Subsystem — NASA Configuration

For USAF missions, the basic philosophy has been to maintain the regular Apollo equipment for the transmission of CSM data (housekeeping and flight control data) to and from NASA ground stations and utilize separate equipment for the transmission of experimental data to and from USAF ground stations. This experimental data would be on USAF frequencies, band-widths, and information bit rates, which are different from those used by NASA ground stations. The AF communications sytem is shown in Figure 3.

The USAF equipment also can be divided into data, radio, and antenna equipment. The data equipment includes a signal processor, unscrambler, and command decoder in the up-link, and a signal conditioner, dual PCM telemetry and scrambler, and a premodulation processor in the down-link. This equipment might also include a timer and data recorder if required. The radio equipment includes an S-band receiver, a coherent S-band transmitter, and two noncoherent S-band transmitters. The antenna equipment includes an S-band multiplexer and three omnidirectional S-band antennas, flush-mounted if possible.

These different equipments, for both NASA and USAF configurations, are described in detail in the following sections. The ground network operations also are discussed. The problems and philosophy of ground control operations, were covered in previous studies (SID 64-1860-20), and are not considered here; only the ground coverage aspects are presented.

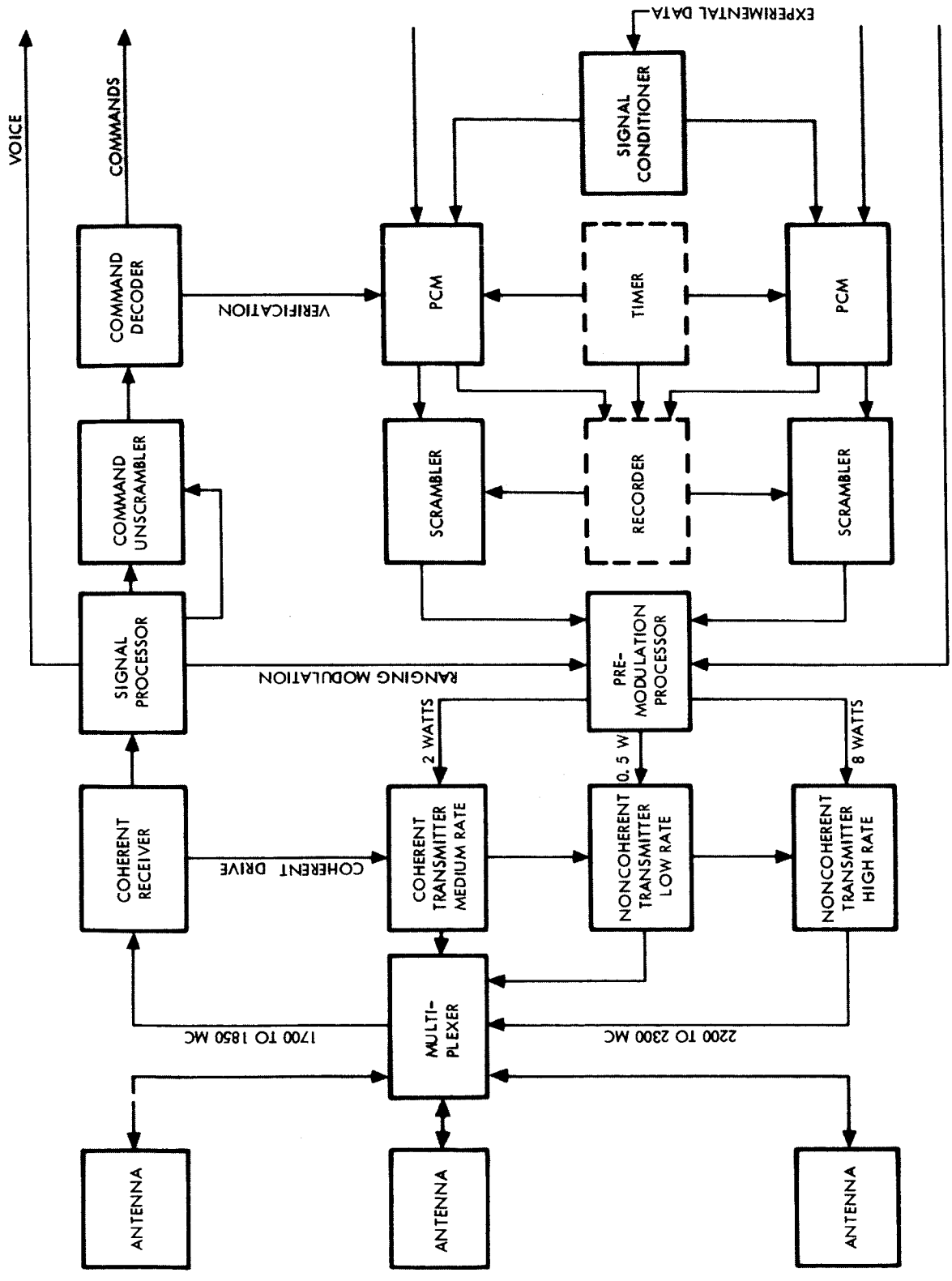

Figure 3. Communications and Data Subsystem — Air Force Configuration

GROUND NETWORK OPERATIONS

Consistent with the approach discussed in the introduction of this volume, NASA and AF flights are presented separately. Because NASA is the only agency experienced in manned spacecraft operations, the general ground coverage aspects are based on NASA operational philosophy. It is assumed that AF operational philosophy will be similar to that of NASA.

A ground network includes a group of communication and tracking stations linked with a control center such as NASA's. The Manned Spacecraft Control Center. The network plus the control center provides the capability for flight control experimental data reception and control of experiments. It is anticipated that these flight control support requirements required in Apollo and Gemini will also be imposed upon the Apollo Extension Systems missions. Flight control support imposes the following network functions on the network:

1. Tracking
2. Voice communication with spacecraft.
3. Monitoring and evaluation of spacecraft systems and crew status at least once every orbit
4. Monitoring and evaluation of experimental systems
5. Computing and evaluating spacecraft trajectory
6. Providing updating data for the vehicle systems

Functional requirements are imposed on the network and the control center during the various mission phases. Tracking is required throughout the ascent phase and insertion into orbit. Tracking and system checks are continued for the first few orbits to verify this achievement. System checks and tracking are required prior to deorbit and through the deorbit and reentry phases up to spacecraft recovery.

NASA EXPERIMENTAL MISSIONS

The Gemini/Apollo network was evaluated in terms of adequate coverage for flight control-housekeeping data and experimental data. The ground stations and their capabilities are listed in Table 3. Included in this table are the new S-band stations under construction and the Madagascar Station, which is in the planning stage.

Table 3. Ground Station Capabilities

Station Name and No.		Location		Tracking		Telemetry		Voice		Command S-band	TV S-Band	
		Latitude (Degrees)	Longitude, East (Degrees)	C-Band	S-Band	S-Band	VHF-FM	S-Band	VHF-AM			
Kano, Africa	801	12 N	8						X			
Carnarvon, Australia	803	25 S	114	X	X	X	X	X	X	X	X	Flight control
Canberra	804	35 S	148		X	X		X		X	X	Deep space flight control
Guam	805	12 N	145		X	X		X		X	X	Flight control
Canton Island	806	2 S	188						X			
Kauai	807	22 N	200	X	X	X	X	X	X	X	X	Flight control
Point Mugu Point Arguello	808	35 N	239	X	X		X		X			
Guaymas	810	28 N	248		X	X	X	X	X	X	X	Flight control
White Sands	811	32 N	254						X			
Corpus Christi, Texas	812	28 N	263		X		X		X			
Cape Kennedy	814	28 N	279	X	X	X		X	X	X	X	Flight control
Wallops Island	815	37 N	284		X	X		X	X	X	X	Training station
San Salvador	816	24 N	285	X					X			
Grand Turk Island	817	21 N	288		X				X			

Table 3. Ground Station Capabilities (Cont)

Station Name and No.	Location		Tracking		Telemetry		Voice		Command S-band	TV S-Band	
	Location (Degrees)	Longitude, East (Degrees)	C-Band	S-Band	S-Band	VHF-FM	S-Band	VHF-AM			
Bermuda 818	33 N	297	X	X	X		X		X	X	Flight control
Canary Island 820	28 N	344		X	X	X	X	X	X	X	Flight control
Madrid 822	37 N	355		X	X		X		X	X	Deep space
Madagascar 323	18 S	48		X	X		X	X	X	X	Flight control

Ground station flight control capability is dependent upon the availability of on-site computers and flight controllers to make mission decisions. Stations with such a capability include Carnarvon, Canberra, Guam, Kauai, Guaymas, Corpus Christi, Cape Kennedy, Bermuda, Canary Islands, and Madagascar.

As presently envisioned, the time constant for ground command reaction to experiment telemetry data will not need to be less than one orbit period (approximately 90 minutes for low altitude missions). No tight closed-loop operations are planned for NASA experiments. The spacecraft rf equipment is to be time-shared and always available for experiment data transmission, except for the approximately 4 to 7 minutes per orbit required for flight control. Data rates, formats, and subcarriers for experimental data are the same as those for Apollo; therefore, ground network provisions for rf transmission/reception, subcarrier generation/reception, and telemetry or command data-handling remain the same as do those for the Apollo mission with no equipment modifications.

Real time telemetering of data from experiments performed during spacecraft ground communications access permits limited high-rate ground recovery of data. Most of the experiment data will be recorded during orbit and dumped with high data rate transmission during communication access.

The estimated NASA ground station coverage for VHF-AM and S-band are shown in the following tables for all the mission orbital inclinations (all at 200-nautical-mile altitude). Coverage was calculated for 100 orbits and a typical slice of 45 orbits is shown. Orbit number one starts at Cape Kennedy for all tables.

Inclination (Degrees)	Table
28.5	4
Polar	5
81.5	6
96.5	7
50	8

For the synchronous orbit missions, DSIF station coverage is shown in Figure 4.

For the low inclination orbits (Table 4) there is adequate coverage to satisfy the flight control requirements during each orbit. For the higher inclination orbits (Tables 5, 6, 7, and 8), at least five orbits out of the 45 shown do not provide the required flight control capability. This situation can be alleviated by additional ground stations, tracking ships, or by relaxation of the requirement.

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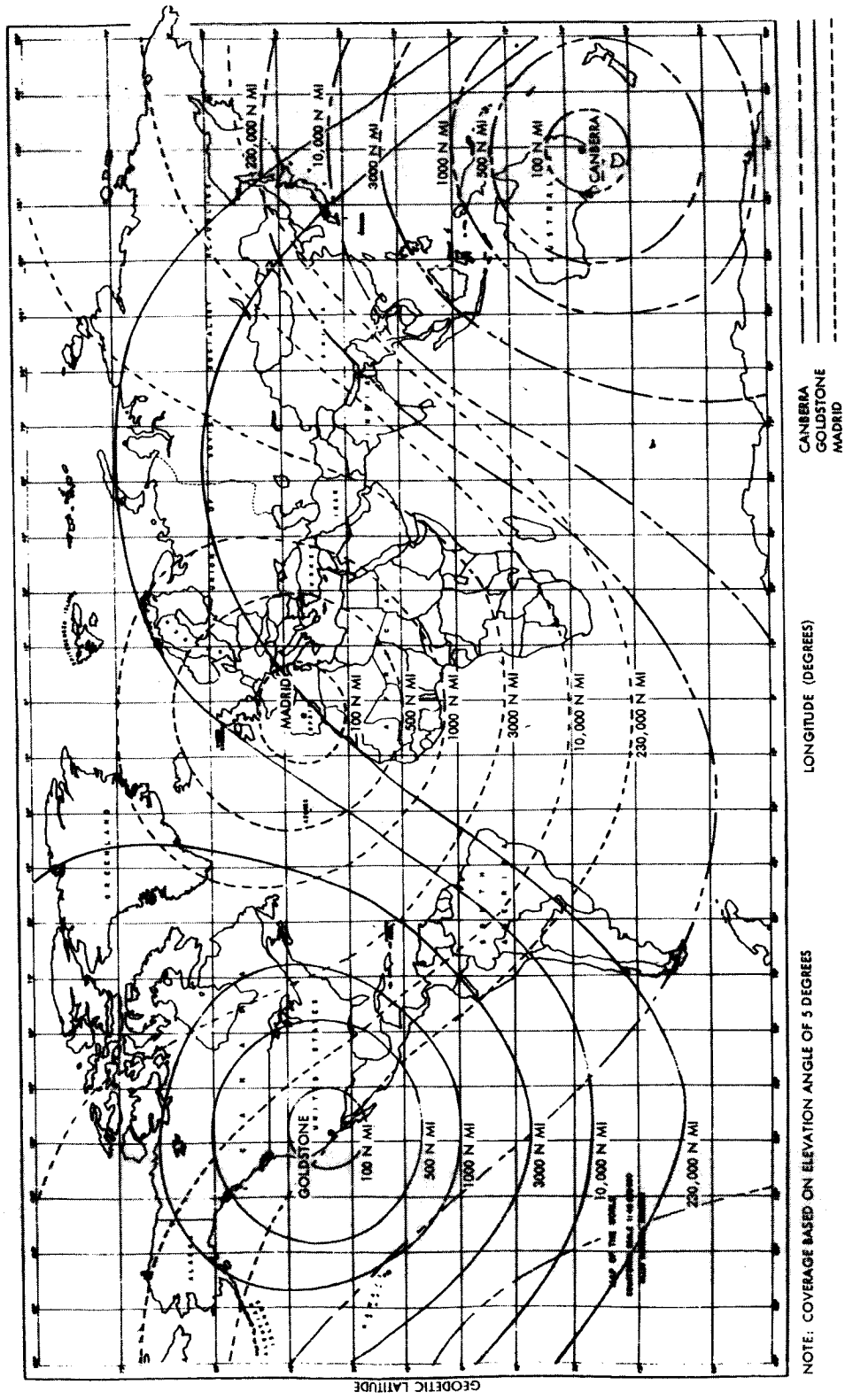

Figure 4. DSIF Station Coverage

Table 4. Ground Station Coverage: 200-Nautical-Mile Altitude 28.5-Degree Inclination, Time in Minutes

Stations \ Orbit		1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23
VHF-AM (voice)	Kano	6.6										6.0			3.0	6.7	6.9							
	Canton Island	7.4															6.9							
	Point Mugu Point Arguello	2.9	6.2	6.8	6.1													5.6	6.7	6.6	4.8			
	White Sands Corpus Christi	7.4	7.5	7.4	6.5											5.7	7.4	7.5	7.5	7.1				
	San Salvador Grand Turk Island	6.5	7.1	7.4										4.3	7.4	7.2	6.6	6.8	7.5	6.6				
S-band (tracking, telemetry, TV, voice, command)	Canberra													5.0	6.6	6.7								
	Guam			4.2	7.4	6.3	1.9			6.1 (FC)	7.5 (FC)									7.0	7.1	4.2		
	Bermuda	7.1	6.1												4.8	6.8	7.1	6.7						
	Madrid												4.3	6.0	5.9									
S-band and VHF-AM	Cape Kennedy	7.5	7.5	7.0											4.1	7.1	7.5 (FC)	7.5 (FC)	7.4 (FC)	5.6				
	Wallops Island	6.1	5.5													4.2	6.0	5.9						
	Canary Islands											4.1 (FC)	6.8 (FC)	7.5 (FC)	7.5 (FC)	7.4 (FC)	6.3							
	Madagascar	7.5	7.9	5.5	4.8	7.0	7.4											7.3	5.9	5.3	6.5			
	Carnarvon	7.3	7.5												7.2	7.5	7.3	7.4	7.2					
	Kauai		6.6	7.5	6.9	6.8 (FC)	6.8 (FC)	7.0 (FC)	6.1 (FC)									5.3	7.5	7.2 (FC)	6.8 (FC)	7.2 (FC)	7.5 (FC)	6.9 (FC)
	Guaymas	7.1 (FC)	7.5 (FC)	7.5 (FC)	7.3 (FC)												6.6	7.5	7.5	7.5	6.6			

Station \ Orbit		24	25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40	41	42	43	44	45
VHF-AM voice)	Kano		7.3	6.9	3.4			5.8	7.5										7.4	5.1			4.3
	Canton Island		7.1						5.6	7.5													
	Point Mugu Point Arguello									4.4	6.5	6.7	5.8										
	White Sands Corpus Christi							2.2	7.1	7.5	7.5	7.4	5.8										
	San Salvador Grand Turk Island						6.9	7.4	6.7	6.6	7.3	7.3										5.7	7.5
S-band tracking, telemetry, TV, voice, command)	Canberra					3.3	6.3	6.8	6.1													5.7	6.7
	Guam	6.3 (FC)	7.5 (FC)	6.3 (FC)								5.8	7.5	5.6				6.2 (FC)	7.3 (FC)				
	Bermuda						2.3	6.4	7.1	7.0	5.6												5.7
	Madrid				2.1	5.7	6.1	6.6													5.0	6.1	5.6
S-band and VHF-AM	Cape Kennedy							6.5	7.5 (FC)	7.5 (FC)	7.5 (FC)	6.6											5.4
	Wallops Island							1.6	5.6	5.7	5.0												
	Canary Islands				6.0 (FC)	7.4 (FC)	7.5 (FC)	7.5 (FC)	7.0											4.2 (FC)	7.1 (FC)	7.5 (FC)	7.5 (FC)
	Madagascar								6.0	7.5	4.9	5.3	5.9	7.3									
	Carnarvon						6.5	7.5	7.3	7.3	7.4											5.2	7.4
	Kauai									1.3	7.1	7.4		6.9 (FC)	7.5 (FC)	6.8 (FC)	5.7 (FC)						
	Guaymas								5.6	7.3	7.5	7.5 (FC)	7.1 (FC)										

(FC) - FLIGHT CONTROL VHF - AM (AVG) = 5.22 hours per day S-band (average) = 3.96 hours per day

Table 5. NASA Ground Station Coverage: 200-Nautical-Mile Altitude, Polar Inclination, Time in Minutes

Stations \ Orbit		1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23
VHF-AM (voice)	Kano					7.5								4.3										
	Canton Island																							
	Point Mugu Point Arguello			7.4							5.1								3.6	6.1				
	White Sands Corpus Christi		7.2							5.2								4.1	5.1					
	San Salvador Grand Turk Island								5.6								6.4	2.9						
S-band (tracking, telemetry, TV, voice Command)	Canberra						3.8								4.1								6.8 (FC)	
	Guam							7.5 (FC)								2.5 (FC)							3.6	5.8 (FC)
	Bermuda	4.3				5.9			7.5 (FC)								7.4 (FC)							
	Madrid			6.7 (FC)											5.8							7.3		
S-band and VHF-AM	Cape Kennedy	7.4 (FC)								7.3 (FC)								7.1 (FC)						
	Wallops Island	7.4							5.1								5.3	6.5						
	Canary Islands						7.5 (FC)								7.5 (FC)							2.9 (FC)		
	Madagascar																			7.5 (FC)				
	Carnarvon																7.4							
	Kauai				4.9 (FC)	5.5 (FC)							6.8 (FC)								7.3 (FC)			
	Guaymas		5.9 (FC)	4.9							7.3 (FC)								7.4 (FC)					

Station \ Orbit		24	25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40	41	42	43	44	45
VHF-AM (voice)	Kano						4.6							6.5									
	Canton Island					4.4																7.1	
	Point Mugu Point Arguello			7.2								7.3								7.4			
	White Sands Corpus Christi		7.3								7.4								7.4				
	San Salvador Grand Turk Island	7.5								7.5								7.5					
S-band (tracking, telemetry, TV, voice, command)	Canberra							6.8															7.4
	Guam	7.1 (FC)						6.2								7.0 (FC)							7.4 (FC)
	Bermuda								2.7 (FC)	7.0 (FC)							6.3 (FC)						
	Madrid						7.4								7.4							7.3	
S-band and VHF-AM	Cape Kennedy	5.4								6.0	2.9 (FC)							7.5 (FC)					
	Wallops Island	7.1								7.1	2.7							7.5					
	Canary Islands						4.0 (FC)	6.4 (FC)							6.6 (FC)							6.9 (FC)	1.4
	Madagascar											2.2											
	Carnarvon																						
	Kauai					7.5 (FC)								7.4 (FC)			5.0				3.2 (FC)		
	Guaymas			7.4 (FC)								7.2 (FC)											

(FC) - Flight Control VHF-AM (average) = 1.51 hours per day S-band (average) = 0.99 hours per day

Table 6. NASA Ground Station Coverage: 200-Nautical-Mile Altitude, 81.5-Degree Inclination, Time in Minutes

Stations \ Orbit		1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23
VHF-AM (voice)	Kano																				5.9			
	Canton Island																				5.2			
	Point Mugu Point Arguello			7.5							7.3								5.6	6.1				
	White Sands Corpus Christi		7.3							7.2								6.1	4.7					
	San Salvador Grand Turk Island	6.6							7.3								6.8							
S-band (tracking, telemetry, TV, voice, command)	Canberra						6.9 (FC)								4.1									
	Guam							7.1 (FC)							5.5								5.8 (FC)	3.9
	Bermuda	4.9							7.1								7.5 (FC)							6.6 (FC)
	Madrid					7.4		2.7							6.0						3.0	7.2		
S-band and VHF-AM	Cape Kennedy	7.5 (FC)							5.4 (FC)									7.0 (FC)						
	Wallops Island	7.5							7.3								5.1	6.6						
	Canary Islands					2.7									7.5 (FC)							6.8 (FC)		
	Madagascar			7.0 (FC)																7.4 (FC)				
	Carnarvon																7.5							5.2
	Kauai				5.4 (FC)	5.1 (FC)							7.5 (FC)								7.4 (FC)			
	Guaymas		6.0 (FC)	5.0							7.4 (FC)								7.5 (FC)					

Stations \ Orbit		24	25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40	41	42	43	44	45
VHF-AM (voice)	Kano						2.7																
	Canton Island					6.6																7.5	
	Point Mugu Point Arguello			7.3								7.3							6.2				
	White Sands Corpus Christi		7.3								7.5							5.8					
	San Salvado Grand Turk Island	7.4								7.5							5.8						
S-band (tracking, telemetry, TV, voice, command)	Canberra							7.2							5.1								
	Guam							7.5 (FC)								7.5 (FC)							
	Bermuda								3.7 (FC)	6.8							7.5 (FC)						
	Madrid						7.5							6.5								7.2 (FC)	
S-band and VHF-AM	Cape Kennedy	7.4 (FC)								6.6 (FC)	3.9 (FC)							7.2 (FC)					
	Wallops Island	7.4								7.2	2.9						6.1						
	Canary Islands						5.1 (FC)	5.9							7.5 (FC)								7.3 (FC)
	Madagascar			5.7 (FC)								5.1 (FC)											
	Carnarvon								1.9														
	Kauai				3.4 (FC)									7.0 (FC)							7.0 (FC)		
	Guaymas		5.6 (FC)								1.7	6.9								7.4 (FC)			

(FC) = Flight Control VHF - AM (average) = 1.57 hours per day S-band (average) = 1.10 hours per day

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Table 7. NASA Ground Station Coverage: 200-Nautical-Mile Altitude, 96.5-Degree Inclination, Time in Minutes

Stations \ Orbit		1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23
VHF-AM (voice)	Kano					7.2																7.5		
	Canton Island													7.5										
	Point Mugu Point Arguello			7.3								7.4							5.8	6.3				
	White Sands Corpus Christi		7.2								7.3							4.9	5.2					
	San Salvador Grand Turk Island	6.9								7.4							5.4	4.9						
S-band tracking, telemetry, TV, voice command)	Canberra														3.9									
	Guam							7.5 (FC)								7.5 (FC)								7.0 (FC)
	Bermuda	3.9							6.3								7.5 (FC)							
	Madrid						7.4								5.9							5.1		
S-band and VHF-AM	Cape Kennedy	7.5 (FC)								7.4 (FC)								7.4 (FC)						
	Wallops Island	7.4								7.5							5.1	6.6						
	Canary Islands						6.9 (FC)								7.5 (FC)								7.5 (FC)	
	Madagascar			6.4 (FC)								6.5 (FC)								7.5 (FC)				
	Carnarvon								5.6 (FC)								7.2							
	Kauai				4.3 (FC)	6.0 (FC)							4.2 (FC)								6.6 (FC)	1.8 (FC)		
	Guaymas		5.9 (FC)	5.1							5.6 (FC)								7.2 (FC)					

Stations \ Orbit		24	25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40	41	42	43	44	45
VHF-AM (voice)	Kano																						4.9
																						4.3	
	Point Mugu Point Arguello			4.0								6.9	3.8							6.4			
	White Sands Corpus Christi		4.5								6.9	2.1							6.7				
	San Salvador Grand Turk Island	5.5									2.9							7.1					
S-band (tracking, telemetry, TV voice command)	Canberra							6.3								5.6	5.1 (FC)						
	Guam															4.8							
	Bermuda	7.3 (FC)								7.3 (FC)													
	Madrid							2.6										7.5					7.5
S-band and VHF-AM	Cape Kennedy		7.5 (FC)							4.4	5.3 (FC)							3.7 (FC)					
	Wallops Island	3.7	7.0							6.8	4.4							8.3					
	Canary Islands							7.1 (FC)								7.2 (FC)							5.5 (FC)
	Madagascar												7.4 (FC)		6.8 (FC)								
	Carnarvon									7.5													
	Kauai					6.6 (FC)								7.5 (FC)								7.5 (FC)	
	Guaymas			7.1 (FC)								7.5 (FC)								7.5 (FC)			

(FC) = Flight Control VHF-AM (average) = 1.57 hours per day S-band (average) = 1.10 hours per day

Table 8. NASA Ground Station Coverage: 200-Nautical-Mile Altitude, 50-Degree Inclination, Time in Minutes

Stations \ Orbit		1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20	21	22	23
VHF-AM (voice)	Kano			6.6																7.4				
	Canton Island			2.2																				
	Point Mugu Point Arguello			7.4	6.3	2.1	1.4	6.0	7.5										6.0	7.1			4.4	7.3
	White Sands Corpus Christi	4.4	7.4	1.1				7.3										7.3	5.7	7.3	4.4		5.9	7.2
	San Salvador Grand Turk Island	5.9					6.4									2.4	7.4					2.5	7.5	
S-band (tracking, telemetry, TV, voice, command)	Canberra			5.5	7.5										4.8	7.5 (FC)	5.1		3.5	7.0				
	Guam						7.4	2.6														5.1	6.8	
	Bermuda	7.3	3.3		1.3	6.8 (FC)	6.9										7.5	5.7			5.2	7.5 (FC)		
	Madrid	4.1	6.6											5.2	7.5		3.5	5.4	7.5					
S-band and VHF-AM	Cape Kennedy	7.5 (FC)	4.8				6.5	6.8 (FC)									6.2	6.9 (FC)				4.0	7.5 (FC)	
	Wallops Island		7.0	4.4	3.9	6.5	7.4										4.9	7.5	5.7	3.5	5.3	7.4		
	Canary Islands			5.6	7.3 (FC)									4.6 (FC)	7.3 (FC)					7.4 (FC)				
	Madagascar		2.1	7.4 (FC)						7.3 (FC)									7.1					
	Carnarvon					2.2										4.3	7.3 (FC)					6.5		
	Kauai				7.3	4.7					7.3 (FC)									5.1	7.1			
	Guaymas		6.8 (FC)	6.4				4.9	7.5 (FC)									2.9	7.4 (FC)	3.0				7.1 (FC)

Station \ Orbit		24	25	26	27	28	29	30	31	32	33	34	35	36	37	38	39	40	41	42	43	44	45
VHF-AM (voice)	Kano											6.9											
	Canton Island											3.9		1.3									
	Point Mugu Point Arguello											7.5	6.1		2.1	6.3	7.5						
	White Sands Corpus Christi									5.2	7.3				2.3	7.5							
	San Salvador Grand Turk Island								7.1	5.3					6.8								
S-band (tracking, telemetry, TV, voice Command)	Canberra							7.3	6.6			5.8	7.5										5.4
	Guam														7.3								
	Bermuda								6.4	7.1	2.6		2.3	7.0 (FC)	6.6								
	Madrid						7.3		4.1	5.9	6.8	7.3 (FC)							5.8			5.8	7.5
S-band and VHF AM	Cape Kennedy									7.5 (FC)	4.2				6.8 (FC)								
	Wallops Island									7.2	6.8	4.3	4.1	6.6	7.3								
	Canary Islands						7.3 (FC)	5.6				6.0	7.1						5.4			5.4 (FC)	7.2 (FC)
	Madagascar	5.4 (FC)						6.8 (FC)			3.8												
	Carnarvon								7.3 (FC)					3.5									
	Kauai		5.5 (FC)	7.1 (FC)									7.4 (FC)	3.9					7.5 (FC)				
	Guaymas										7.4 (FC)	6.1				5.4 (FC)	7.4 (FC)						

(FC) - Flight Control VHF-AM (average) = 3.07 hours per day S-band (average) = 2.86 hours per day

Average voice communication capability (VHF-AM) decreases from 5.22 hours per day for the 28.5-degree orbital inclination to 1.51 hours per day for the polar inclination orbit, while the S-band coverage decreases simultaneously from an average 3.96 hours per day to 0.99 hours per day for the same inclinations.

AIR FORCE EXPERIMENTAL MISSIONS

The space-ground systems (SGLS) spaceborne and ground equipment being developed for AFSC-SSD by Space Technology Laboratories, intended for application to all SSD manned and unmanned programs, was selected with some modification for these missions.

While there is a gross similarity between the NASA spaceborne system and SGLS, in capability at least, differences in rate and format preclude a simple interface between SGLS and the Apollo spacecraft and ground data subsystems. The experiment data/communications subsystem for Air Force experiments will be entirely separate from the spacecraft subsystems, except for power.

For the most part, the Air Force experiments do not impose rapid closed-loop data transfer operations, and a one-orbit period delay between receipt of down-link data and response with up-data commands is acceptable. In a few cases, where communications experiments are scheduled to operate in real time, more rapid response may be necessary. These cases cannot be accurately identified until a better picture of the Air Force ground network provisions is available.

Equipment presently installed at NASA stations cannot process SGLS data rates and formats. SGLS ground data-handling equipment must be provided to handle this traffic. Therefore, the Air Force experiment data will be transmitted to Air Force ground stations, while flight control for the Apollo vehicle will remain with the Gemini/Apollo network.

The Air Force ground stations identified in the MOL RFP were evaluated in terms of adequate coverage for experimental data and control for both secure and nonsecure stations. Flight control evaluation was based on the Gemini/Apollo network and its compatibility with the Apollo onboard systems. The estimated ground station coverage is shown in Table 9. Coverage was calculated for 100 orbits, and a typical slice of 45 orbits is shown. Orbit 1 starts at Cape Kennedy.

It is apparent from Table 9 that flight control coverage is never less than 4.1 minutes for an orbit and that coverage for 43 of the 45 orbits shown is never less than 6 minutes. The daily average of 1.65 hours per day secure communication and 0.95 hours per day of nonsecure communication more than meet the mission requirements.

Table 9. Air Force Ground Station Coverage 200-Nautical-Mile Altitude, 28.5-Degree Inclination, Time in Minutes

Stations	Orbit	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
Secure Stations	Okinawa						6.2	7.5	7.5	6.8	4.1					
	Hawaii		6.6	7.5	6.9	6.8	6.8	7.0	6.1							
	PMR	2.9	6.2	6.8	6.1											
	Antigua	5.1	6.8	7.5										7.2	7.1	5.6
Non-secure stations	CKAFS	7.5	7.5	7.0											4.1	7.1
	Carnarvon	7.3	7.5												7.2	7.5
Flight control	Guam									6.1	7.5					
	Cape Kennedy															
	Canary Islands											4.1	6.8	7.5	7.5	7.4
	Kauai					6.8	6.8	7.0	6.1							
	Guaymas	7.1	7.5	7.5	7.3											

Stations	Orbit	16	17	18	19	20	21	22	23	24	25	26	27	28	29	30
Secure Stations	Okinawa						2.7	5.9	7.4	7.5	6.1					
	Hawaii		5.3	7.5	7.2	6.8	7.2	7.5	6.0							
	PMR		5.6	6.7	6.6	4.8										
	Antigua	4.7	6.1	7.5	6.5									6.3	7.5	6.2
Non-secure stations	CKAFS	7.5	7.5	7.4	5.6											6.5
	Carnarvon	7.3	7.4	7.2											6.5	7.5
Flight control	Guam									6.3	7.5	6.3				
	Cape Kennedy															
	Canary Islands	6.3											6.0	7.4	7.5	7.5
	Kauai						7.2	7.5	6.9							
	Guaymas		7.5	7.5	7.5	6.6										

Stations	Orbit	31	32	33	34	35	36	37	38	39	40	41	42	43	44	45
Secure Stations	Okinawa							6.1	7.3	7.4	7.5	6.4				
	Hawaii		1.3	7.1	7.4	7.5	6.9	7.5	6.8	5.7						
	PMR		4.4	6.5	6.7	5.8										
	Antigua	4.8	5.5	7.1	7.3									4.3	7.5	6.8
Non-secure Stations	CKAFS	7.5	7.5	7.5	6.6											
	Carnarvon	7.3	7.3	7.4											5.2	7.4
Flight control	Guam										6.2	7.3				
	Cape Kennedy															
	Canary Islands	7.0											4.2	7.1	7.5	7.5
	Kauai						6.9	7.5	6.8	5.7						
	Guaymas		7.3	7.5	7.5	7.1										

Note: Secure Stations (average) = 1.65 hours per day Nonsecure stations (average) = 0.95 hours per day

DATA TRANSMISSION/RECEPTION EQUIPMENT

This section describes the rf equipment used for the reception and transmission of experimental and housekeeping data for both NASA and USAF missions. Again, there are two different configurations that are considered.

The NASA missions use the Apollo system for both housekeeping and experimental functions. Experimental and housekeeping functions use this equipment at different times during the mission; the time difference is due to ground access and the fact that one particular station per orbit is used for flight control. Because the Apollo equipment is well treated in other documents, such as SID 64-1860-18, it will not be considered here.

The Air Force missions use the Apollo system for housekeeping functions and use a separate independent system for experimental functions. Because this equipment may be unfamiliar, it is discussed in the following paragraphs.

The Air Force equipment provides for the reception of voice, command, and tracking signals from the ground and for the transmission of voice, telemetry, and tracking signals to the ground.

The equipment, shown in a block diagram in Figure 3, comprises a coherent S-band receiver, a coherent S-band, two noncoherent S-band transmitters, an S-band multiplexer, and three omnidirectional antennas, flush-mounted if possible. This equipment operates on frequencies, bandwidths, data rates and formats that are different from those used for NASA missions. There is a possibility, however, that the same S-band omnidirectional antennas could be used with both NASA and USAF radio equipment if frequency-independent antennas, such as scimitars, spirals, and conical helices, were used. These different devices are described separately.

A coherent phase-lock receiver with a 205/8 receiver-to-transmitter drive ratio is used. The receiver is capable of receiving any fixed frequency signal in the 1762 to 1842 megacycles range through selection of the VCO frequency. This receiving band covers the entire allocations by translation into a 2200 to 2300 megacycles transmitting band.

The block diagram of Figure 5 shows the synthesis of the 205-to-8 ratio. The received signal at 205 F_1 is heterodyned in the first mixer with the local oscillator signal at 200 F_1, to give an output at 5 F_1. The output of the first IF amplifier at 5 F_1 is heterodyned in the second mixer with a

signal at 5 F_1-F_2 to give an output at F_2. The signal at F_2 is amplified and band-limited in the second IF amplifier and applied to the phase detector that drives the operational amplifier, which, in turn, controls the VCO that operates at a nominal frequency of 2 F_1. The VCO signal is multiplied by 100 to a frequency of 200 F_1 and injected into the first mixer to close the loop.

In addition, the VCO signal is multiplied by five and mixed with the reference oscillator signal at 2 F_2 to give an output at 10 F_1 - 2 F_2. This signal is divided by two to generate the required 5 F_1 - F_2 signal at the second mixer. The reference oscillator signal also is divided by 2 to give an output at F_2, which is used as the reference signal for the phase detectors. The feedback to the VCO controls its frequency and phase and thus the frequency of each mixer injection signal is maintained in precisely the relation indicated previously. In addition, the IF and reference injections at the loop phase detector are very nearly in phase quadrature. The output of the VCO at 2 F_1 is subsequently multiplied by four, giving the 8 F_1 transmitter drive at approximately 72 megacycles.

The second local oscillator signal is obtained by mixing and frequency division rather than by operating the VCO at the second local oscillator frequency. This procedure obviates the need for a mixer in or before the first local oscillator multiplier chain, and thus the possibility of generation or enhancement of nonharmonically related spurious products is eliminated from such a mixer in the frequency multiplier chain to the transmitter and first local oscillator.

This particular means of synthesizing the coherent ratio of 256 to 205 achieves two important objectives. First, no high-level subharmonics of the received frequency or first IF frequency are generated either as local oscillator signals or as mixer products. These subharmonics are avoided through the use of the double conversion technique. If products of this nature were generated, the receiver would tend to lock on itself, seriously impairing its performance in this normal mode. Secondly, this particular block diagram (Figure 5) permits the selection of frequencies such that all circuitry and devices are operating conservatively. For example, the first IF is approximately 45 megacycles, the second IF and phase detector are operating at 12.5 megacycles, and the reference oscillator and VCO are operating at 25 and 18 megacycles, respectively. In addition, providing the transmitter drive at 72 megacycles permits the generation of the required high power level, from 10 to 12 watts, at a conservative 2 x 72 = 144 megacycles.

There are three S-band transmitters: one is phase-locked to the receiver, and two are independent and could be phase-locked if required. The coherent transmitter provides a power output level of 500 milliwatts, while the two independent transmitters can provide 2 watts and 8 watts,

respectively. The 500-milliwatt transmitter is used to transmit low-rate data, the 2-watt transmits medium-rate data, and the 8-watt transmitter is used to transmit high-rate data.

The 2-watt transmitter, diagrammed in Figure 6, operates as follows. A diode switch selects the output of the crystal oscillator or coherent drive from the receiver. The coherent drive signal is at a frequency of 8 F_1, (about 70 megacycles), and the level is 3 dbm ±3 dG. The 8 F_1 signal is doubled and used to drive a phase modulator operating at 16 F_1. Four cascaded stages of power amplification drive a chain of four cascaded varactor doublers that generate the transmitter output frequency of 256 F_1 = 2240 megacycles.

The two other transmitters, 5 megacycles below and 5 megacycles above the medium-rate transmitter, can operate from a self-contained crystal oscillator or from the receiver. When they are used as coherent transmitters mode switching from noncoherent to coherent operation is controlled by the signal-present circuits in the sweep acquisition portion of the receiver.

The multiplexer consists of a conventional combination of ferrite circulators, band-pass and band-reject filters, and hybrid networks. The multiplexer is used to combine or separate the transmitter outputs and distribute the resulting output between the antennas or connect the antennas to the receiver.

Omnidirectional S-band antennas do not present any particular problem of size, weight, or radiation pattern. Their particular physical configuration will depend upon the size and shape of the external device. These antennas will be preferably flush-mounted and possibly of a frequency-independent type, such as spirals, scimitars, or conical helices.

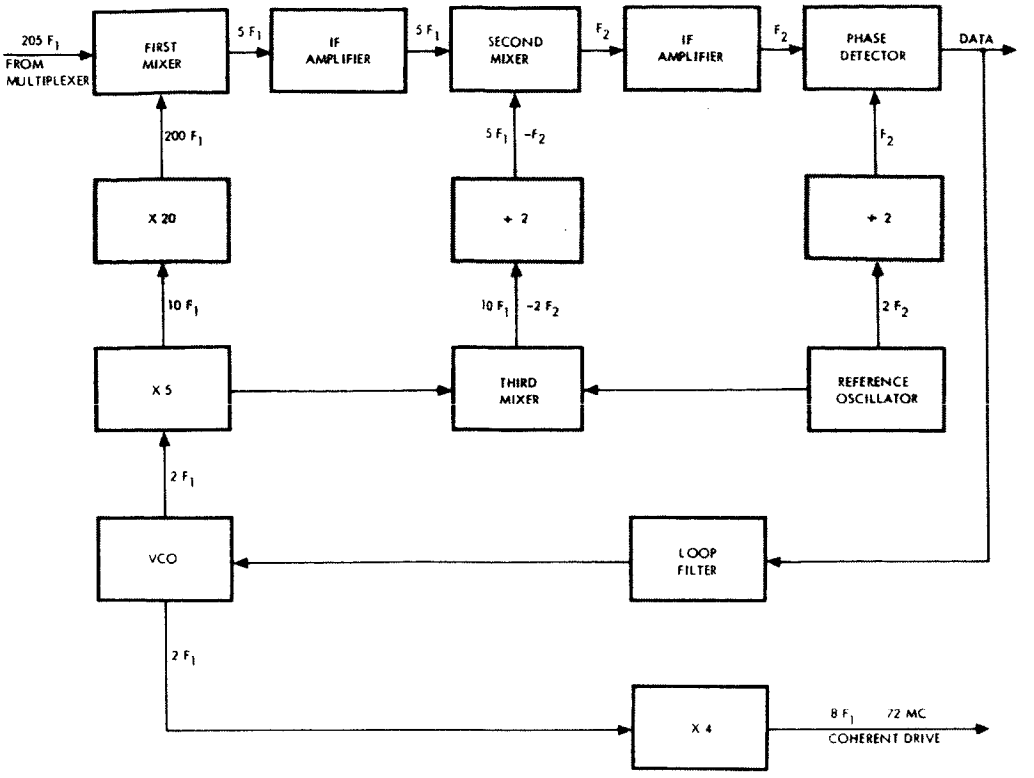


Figure 5. Air Force S-Band Receiver

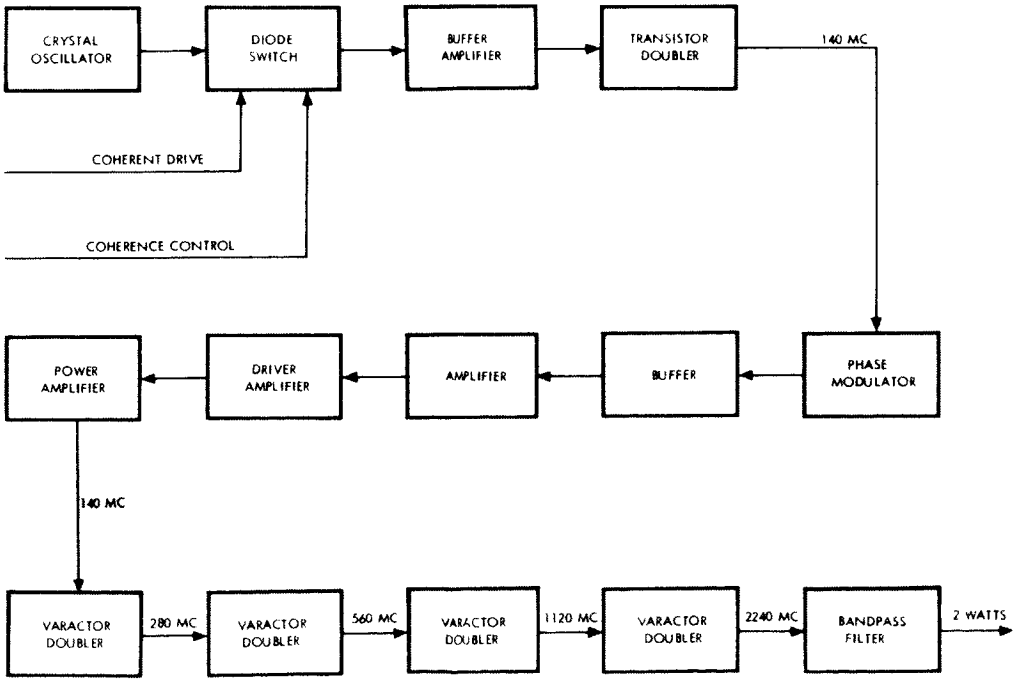


Figure 6. Air Force S-Band Transmitter

DATA SYSTEMS

The data systems described in this section provide means for returning data from spaceborne packages to Earth and for decoding and implementing ground-originated commands.

The Apollo system conditions, formats, and converts subsystem status data. In the same way, experimental data must be conditioned, formatted, and sometimes converted before transmission. The Apollo data system is not treated here; only the data system for experiments and any status data concerning additional subsystems on an external device are considered.

A general philosophy has been followed for the design of these data systems. Standard data formats and rates have been adopted that satisfy the great bulk of experiment data transmission and reception requirements for all missions; these characteristics are standardized from the point of view of the ground networks and also of the experiment packages. Thus, a substantial portion of the equipment provided solely for experiment package telemetry and command purposes may be designated as experiment standard, and invariant from mission to mission. Requirements not satisfied by the standard characteristics are met by separate experiment-peculiar equipments provided specifically for the purpose. Descriptions of these experiment-peculiar requirements and subsystems may be found in the report sections devoted to the experiments and missions.

NASA MISSION SYSTEM

Conditions for AES mission operations favor time-sharing of portions of the Apollo telecommunications subsystem. Telemetry, command, and flight control operations will not utilize more than four to seven minutes per orbit for spacecraft telemetry transmission and command reception; although the telemetry and command data equipment will be operational all the time, the rf components will be idle except during this four-minute period. During the remainder of the spacecraft ground-network access period of each orbit, the Apollo unified S-band equipment and antennas could be used to transmit and receive experiment mission data, in Apollo formats and Apollo data rates, to and from NASA-ground stations. In order to accomplish this, an experiment data system similar in many respects to that in the Apollo data subsystems at the Apollo premodulation processor. Figure 7 is a functional block diagram illustrating the spacecraft and experiment system interface.

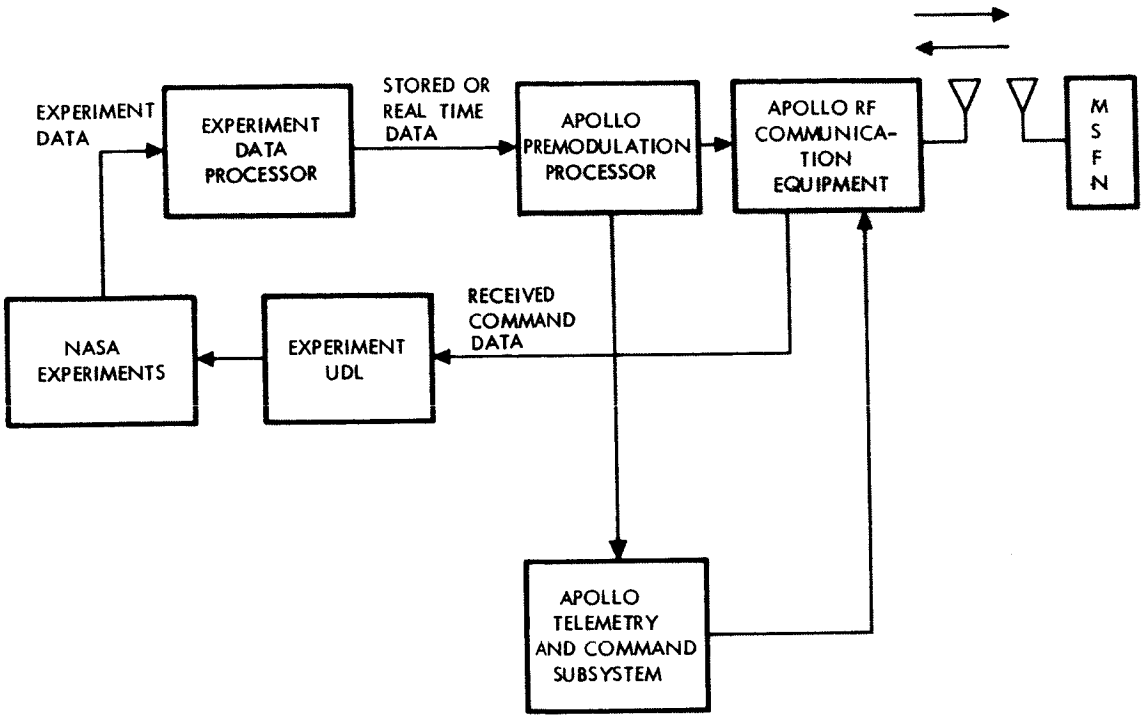


Figure 7. NASA Experiment Data System Interface With Apollo

The experiment data system will be both functionally and physically similar to the existing Apollo data system; Table 10 lists the data acquisition and transmission capabilities this system will provide for experiment performance. Note that the capability shown is entirely independent of that of the spacecraft subsystem. Figure 8 and Table 11 show (for reference only) the high and low data-rate frame formats used in the spacecraft data system.

Up-data link capability for experiment package command from the ground network also duplicates that of the spacecraft data subsystem. The maximum reception rate is five 30-bit words per second, up to a total of 384 words from any one ground station in one orbit.

Figure 9 is a functional block diagram of the experiment data system. The system's interface with the spacecraft communication subsystem is at the premodulation processor, where telemetry and command signals may be routed to or from either the spacecraft data system or the experiment data system. The telemetry interface between the standard experiment system and those components with mission-peculiar characteristics is functionally between the analog-to-digital converter, which converts into digital format all incoming signals in its range, and the analog commutator, which connects analog signals to the converter according to a fixed schedule which is dependent on experiment mission requirements. The command interface is between

the command decoder, which reconstitutes information-bit pulse-code words from the phase-modulated subcarrier received from the S-band receiver, and an experiment command implementation device which will either execute discrete commands in real time or store them for later execution. Brief functional and physical descriptions of both standard data system components, and of Apollo components capable of adaptation for experiment package use, follow. The order in which they are described is that of their functional proximity to the interface with the spacecraft communication subsystem.

Data Storage Equipment

This unit is a magnetic tape recorder which impresses 14 data channels (five digital, nine analog) on a 2400-foot length of one-inch-wide tape. Recording of digital data (PCM-NRZ) is at 1600 bits per second at 3.75 inches per second, or 51,200 bits per second at 15 inches per second; digital data playback is at 51,200 bits per second at 120 inches per second for data acquired at the lower rate, or at 51,200 bits per second at 15 inches per second for data acquired at the higher rate.

Analog data in the frequency range of 12.5 to 5,000 cps may be recorded and played back at 3.75 inches per second. Signals with frequencies below 1250 cps may be recorded at the 3.75-inch per second rate for time-compressed playback at 15 inches per second. The basic analog recording frequency response of 50 to 25,000 cps at 15 inches per second may be employed only if no playback from the spacecraft is required, because transmission modulation frequency response is limited at 5000 cps. Input characteristics for the analog signals entering the recorder are 0 to 5 volts peak-to-peak within the specified frequency range, to an input impedance of 100,000 ohms minimum. The recorder weighs 28.5 pounds maximum, including one reel of tape, is 6 x 9.5 x 22 inches in size, and requires 1.0 watts of 28-volt d-c power and 32 watts of 115-volt, 400-cycle, three-phase power.

PCM Multiplexer

This unit accepts eight-bit, bit-parallel, word-serial data from an analog-to-serial converter, at 6400 or 200 words per second, other parallel and serial digital data inputs, and generates a serial bit stream at either 1600 or 51,200 bits per second (the former rate is expected to be used most on experiment missions).

The input characteristics for parallel digital signals entering the PCM multiplexer directly from the experiments are:

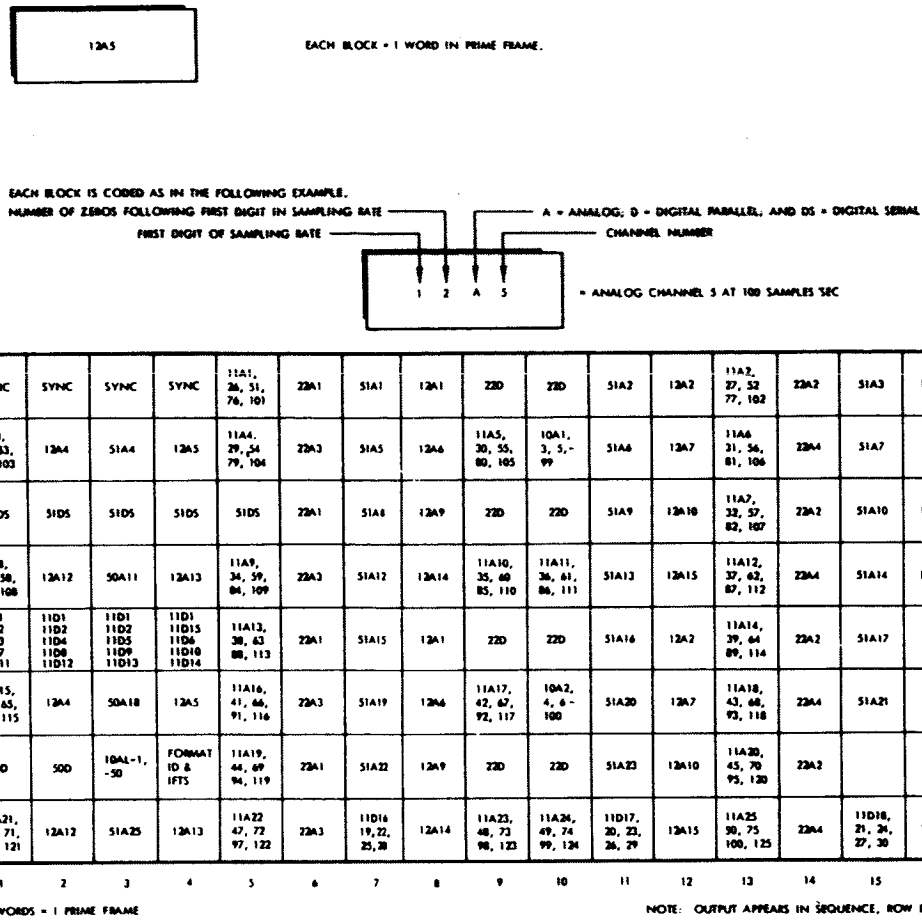
Input level: 0 ± 0.5 v (binary 0), 3.5 to 10 v (binary 1)

Table 10. Data Acquisition, Storage, and Transmission Capability of the NASA Experiment Data System

ANALOG DATA SAMPLED CONVERTED TO DIGITAL PCM-NRZ						
Mode*	Acquisition and Storage			Transmission		
	Rate (kbs)	Time (min)	Speed (ips)	Rate (kbs)	Time (min)	Speed (ips)
1	1.6	120	3.75	51.2	3.75	120
2	51.2	30	15	51.2	30	15
3	51.2	No storage		51.2	Real-time-spacecraft-MSFN access	
ANALOG DATA ACQUIRED AND TRANSMITTED IN ANALOG FORM						
	Frequency Range (cps)	Time (min)	Speed (ips)	Frequency Range (cps)	Time (min)	Speed (ips)
4	12.5 to 5000	120	3.75	12.5 to 5000	120	3.75
5	12.5 to 1250	120	3.75	50 to 5000	30	15
6	12.5 to 5000	No storage		12.5 to 5000	Real-time-spacecraft-MSFN-access	
7	50.0 to 25,000		15.00	Physical return of tape, no playback		

NOTES:

* Modes 1 and 2 are for use in near-earth orbital operations, with Mode 1 expected to be dominant. Mode 3 is for use at greater distances, with spacecraft-ground access of greater duration as in synchronous orbit. There are nine channels available for analog signal recording for a total recording capability of over 19 hours. Analog recording may occur simultaneously with mode 1 (digital recording) in modes 4 and 5. Playback in these modes must be separate from digital playback. Recording in mode 7 must be a separate operation, and data recovery will be by physical recovery of tape and playback on the ground. Mode 6 is analogous to mode 3 and is for use during synchronous orbits.


12A5
EACH BLOCK = 1 WORD IN PRIME FRAME.
CODE: EACH BLOCK IS CODED AS IN THE FOLLOWING EXAMPLE.
NUMBER OF ZEROS FOLLOWING FIRST DIGIT IN SAMPLING RATE
FIRST DIGIT OF SAMPLING RATE
A = ANALOG; D = DIGITAL PARALLEL; AND DS = DIGITAL SERIAL
CHANNEL NUMBER
1 2 A 5
= ANALOG CHANNEL 5 AT 100 SAMPLES SEC
128 WORDS = 1 PRIME FRAME
NOTE: OUTPUT APPEARS IN SEQUENCE, ROW BY ROW.

1	SYNC	SYNC	SYNC	SYNC	11A1, 26, 51, 76, 101	22A1	51A1	12A1	22D	22D	51A2	12A2	11A2, 27, 52 77, 102	22A2	51A3	12A3
2	11A3, 28, 53, 78, 103	12A4	51A4	12A5	11A4, 29, 54 79, 104	22A3	51A5	12A6	11A5, 30, 55, 80, 105	10A1, 3, 5, -99	51A6	12A7	11A6 31, 56, 81, 106	22A4	51A7	12A8
3	51DS	51DS	51DS	51DS	51DS	22A1	51A8	12A9	22D	22D	51A9	12A10	11A7, 32, 57, 82, 107	22A2	51A10	12A11
4	11A8, 33, 58, 83, 108	12A12	50A11	12A13	11A9, 34, 59, 84, 109	22A3	51A12	12A14	11A10, 35, 60 85, 110	11A11, 36, 61, 86, 111	51A13	12A15	11A12, 37, 62, 87, 112	22A4	51A14	12A16
5	11D1 11D2 11D3 11D7 11D11	11D1 11D2 11D4 11D8 11D12	11D1 11D2 11D5 11D9 11D13	11D1 11D2 11D15 11D6 11D10 11D14	11A13, 38, 63 88, 113	22A1	51A15	12A1	22D	22D	51A16	12A2	11A14, 39, 64 89, 114	22A2	51A17	12A3
6	11A15, 40, 65, 90, 115	12A4	50A18	12A5	11A16, 41, 66, 91, 116	22A3	51A19	12A6	11A17, 42, 67, 92, 117	10A2, 4, 6 - 100	51A20	12A7	11A18, 43, 68, 93, 118	22A4	51A21	12A8
7	50D	50D	10AL-1, -50	FORMAT ID & IFTS	11A19, 44, 69 94, 119	22A1	51A22	12A9	22D	22D	51A23	12A10	11A20, 45, 70 95, 120	22A2		12A11
8	11A21, 46, 71, 96, 121	12A12	51A25	12A13	11A22 47, 72 97, 122	22A3	11D16 19,22, 25,28	12A14	11A23, 48, 73 98, 123	11A24, 49, 74 99, 124	11D17, 20, 23, 26, 29	12A15	11A25 50, 75 100, 125	22A4	11D18, 21, 24, 27, 30	12A16
	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16

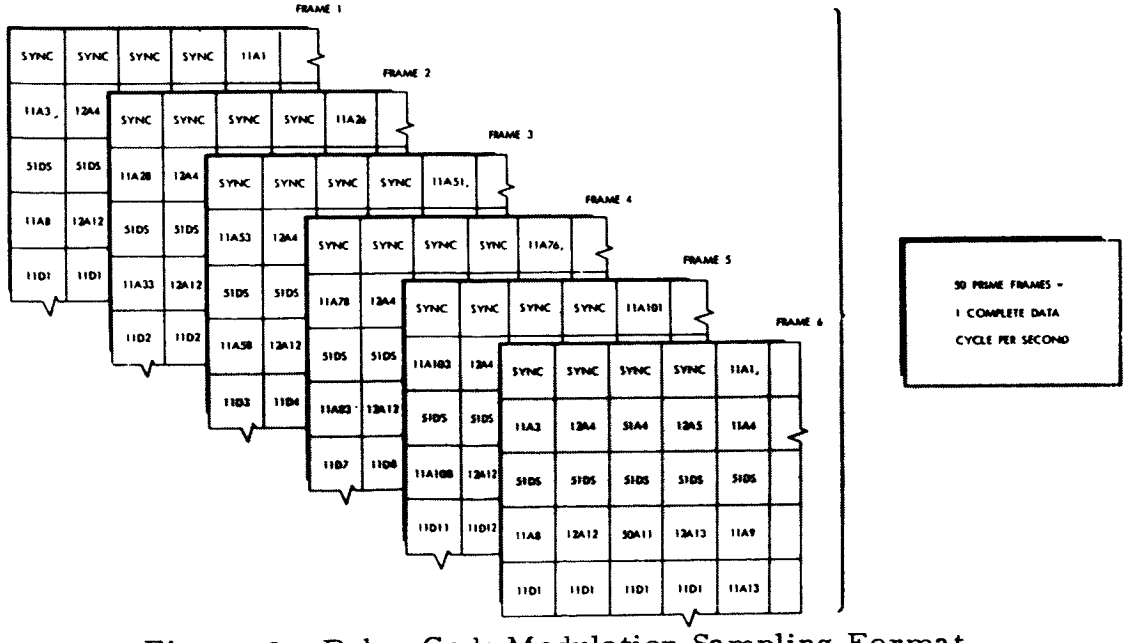


Figure 8. Pulse Code Modulation Sampling Format (Normal Data Rate 51.2 K bps)

Table 11. Pulse Code Modulation Sampling Format (Reduced Data Rate 1.6 K bps)

	1	2	3	4	5	6	7	8	9	10
1	SYNC	SYNC	SYNC	SYNC & ID	10A1	10/11A1	10A3	10/11A2	11/51DS	11/51DS
2	11/51DS	11/51DS	11/51DS	10/11A4	10A6	10AL4	10A7	10/11A6	10A9	10/11A7
3	10/11D3	10/11D4	10/11D15	10/11D16	10A11	10/11A13	10A13	10/11A14	11/51DS	11/51DS
4	11/51DS	11/51DS	11/51DS	10/11A16	10A16	10AL9	10A17	10/11A18	10A19	10/11A20
5	10/11DS	10/11D17	10/11D18		10A21	10/11A26	10A23	10/11A27	11/51DS	11/51DS
6	11/51DS	11/51DS	11/51DS	10/11A29	10A26	10AL14	10A27	10/11A31	10A29	10/11A32
7	10/11D6	10/11D7	10/11D19	OR IFTS(1's)	10A31	10/11A38	10A33	10/11A39	11/51DS	11/51DS
8	11/51DS	11/51DS	11/51DS	10/11A41	10A36	10AL19	10A37	10/11A43	10A39	10/11A45
9	10/11D8	10/11D20	10/11D21	FORMAT ID	10A41	10/11A51	10A43	10/11A52	11/51DS	11/51DS
10	11/51DS	11/51DS	11/51DS	10/11A54	10A46	10AL24	10A47	10/11A56	10A49	10/11A57
11	10/11D1	10/11D1	10/11D1	10/11D1	10A51	10/11A63	10A53	10/11A64	11/51DS	11/51DS
12	11/51DS	11/51DS	11/51DS	10/11A66	10A56	10AL29	10A57	10/11A68	10A59	10/11A70
13	10/11D9	10/11D10	10/11D22	10/11D23	10A61	10/11A76	10A63	10/11A77	11/51DS	11/51DS
14	11/51DS	11/51DS	11/51DS	10/11A79	10A66	10AL34	10A67	10/11A81	10A69	10/11A82
15	10/11D11	10/11D24	10/11D25		10A71	10/11A88	10A73	10/11A89	11/51DS	11/51DS
16	11/51DS	11/51DS	11/51DS	10/11A91	10A76	10AL39	10A77	10/11A93	10A79	10/11A95
17	10/11D12	10/11D13	10/11D26	OR IFTS(0's)	10A81	10/11A101	10A83	10/11A102	11/51DS	11/51DS
18	11/51DS	11/51DS	11/51DS	10/11A104	10A86	10AL44	10A87	10/11A106	10A89	10/11A107
19	10/11D14	10/11D27	10/11D28		10A91	10/11A113	10A93	10/11A114	11/51DS	11/51DS
20	11/51DS	11/51DS	11/51DS	10/11A116	10A96	10AL49	10A97	10/11A118	10A99	10/11A120

ONE FRAME = ONE COMPLETE DATA CYCLE PER SECOND

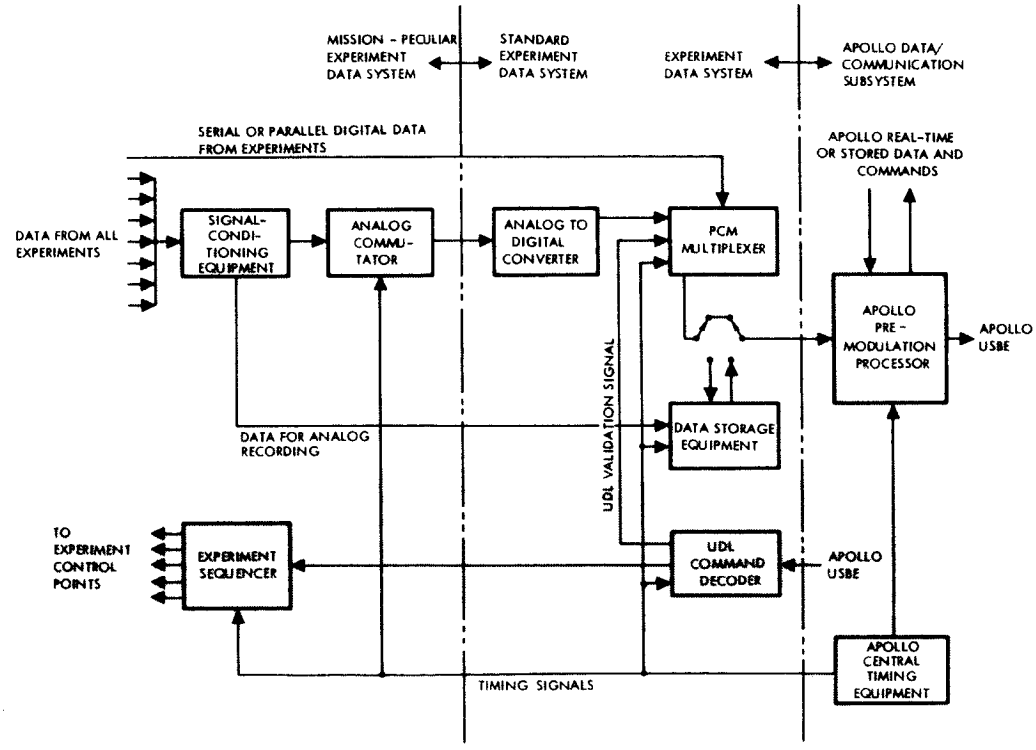


Figure 9. Functional Block Diagram of NASA Experiment Data System

Input impedance 50,000 ohms during sampling, 2 megohms at other times

Source impedance: 5000 ohms maximum

Sampling formats and rates (51.2 kbs operations):

1 channel at 200 samples/second, 16 bits/sample
1 channel at 50 samples/second, 16/bits/sample
1 channel at 10 samples/second, 32 bits/sample
1 channel at 10 samples/second, 24 bits/sample
28 channels at 10 samples/second, 8 bits/sample

Sampling formats and rates (1600 bps operations):

26 channels at 1 sample/second, 8 bits/sample
1 channel at 1 sample/second, 32 bits/sample

For serial digital data inputs:

Input level: 0 ± 0.5 v (binary 0), 3.5 to 10 v (binary 1) RZ

Input impedance: 500 ohms ± 10 percent

Source impedance: 100 ohms maximum

High sampling rate: 1 channel at 50 samples/second, 40 bits/sample

Low sampling rate: 1 channel at 10 samples/second, 40 bits/sample

The input characteristics for digital data signals entering the unit from the analog-to-digital converter are:

Input level: 0 ± 0.5 v (binary 0), 3.5 to 10 v (binary 1)

Input impedance - 50,000 ohms minimum

Source impedance: 5000 ohms maximum

This unit accepts synchronizing signals from the spacecraft central timing equipment. The PCM multiplexer weighs 23 pounds, is 4.2 x 9.4 x 16.5 inches in size, and requires 5 watts of dc power.

Command Decoder

This unit accepts phase-modulated data, impressed on a 70-kc subcarrier, from the S-band receiver. The phase-modulated data are

stripped from the subcarrier and assembled into information bits and words up to 30 bits in length. The output of the decoder includes signals to energize or deenergize up to 32 individual external relays, and serial 30-bit data words, originally intended for entry into a computer.

The output characteristics are:

External Relay Characteristics

d-c coil resistance: 1240 ohms ± 10 percent
coil pickup current: 9 ma dc

Data Work Characteristics

not determined

The command decoder weighs 20 pounds, is 6 x 9.7 x 18.3 inches in size, and requires 15 watts of dc power.

Analog Commutator/Analog-to-Digital Converter

While these functions fall on either side of the standard/mission-peculiar interface postulated earlier, a single unit has been designed for the Apollo Block I spacecraft to perform them. The analog-to-digital conversion performance of the unit is entirely adequate for standard data system use; the commutator accepts normalized analog dc voltage signals, and is properly designed to drive the converter, but its commutation schedule is fixed, and may not be suitable without modification for use with experiment packages. The characteristics of the unit are described herein.

The analog-to-digital converter accepts normalized (0 to 5 volts dc full-scale, unipolar) signals one at a time, from the signal conditioner through the commutator, and encodes them in eight-bit natural binary form for presentation to the PCM multiplexer.

The analog commutator, as provided for Apollo Block I, accepts normalized signals from many different signal conditioner sources, and converts them to the converter according to the following schedule:

For the high rate (51,200 bits/second): 4 channels at 200 samples/second, 16 channels at 100 samples/second, 25 channels at 50 samples/second, 125 channels at 10 samples/second, and 150 channels at 1 sample/second. For the low rate (1600 bits/second): 110 channels at 1 sample/second (i.e., 60 of the 125 high rate channels at 10 samples/second, plus 50 of the 100 high rate channels at 1 sample/second).

Note that the channel and sampling rate figures, plus the direct digital data entry capability, plus frame sync and ID and telemetry validation, add up to the 51,200 and 1600-bit-per-second total capacity. This sum must be retained for any schedule modifications.

The Apollo unit described above weighs 24 pounds, is 4.2 x 9.4 x 16.5 inches in size, and requires 5 watts of dc power.

Signal-Conditioning Equipment

The signal conditioner is hard-wired to all experiments emitting analog data for conversion to PCM, and must therefore be tailored to suit each experiment package. This unit's function is the normalization of all experiment output signals, in level and form, so that each signal may be represented by a dc voltage analog varying from 0 to 5 volts full scale, for presentation in turn to the analog commutator and analog-to-digital converter.

The Apollo unit which performs this function is described for reference purposes. The unit is physically composed of a wired chassis with individual plug-in converter modules for the various signals entering the unit from transducers. Table 12 lists the signal-handling characteristics of the available modules.

This unit with modules weighs 45 pounds, ix 6 x 9.8 x 20.3 inches in size, and requires 18 watts, plus 0.3 watts of dc power per signal processed (about 75 watts total for Apollo).

Experiment Command Sequencer

The Apollo component analogous to the experiment command sequencer is the Apollo guidance computer portion of the guidance and navigation system. Some memory and data-handling capability is required to implement real-time or stored commands and subroutines. No detailed design has been done to determine the characteristics of this unit, but preliminary estimates indicate that physical characteristics would not exceed 10 pounds, 0.25 cubic foot, and 10 watts of dc power.

Subsystem Status Monitoring

In spacecraft configurations employing the external device (D and D'), the presence of fuel cells and cryogenic tanks necessitates monitoring of various parameters, and telemetry to the ground of acquired data. The arrangement shown in Figure 10 avoids proliferation of wires connecting the monitor sensors to the Apollo data subsystem, and use of the experiment data system for monitoring.

Table 12. Signal-Conditioning Equipment Module Characteristics

Module Type	Input Form and Range	Output
01 DC Amplifier	0 to 20 mv dc and all intermediate ranges through 250 mv	0 to 5 vdc, accurate within ± 1.0%
02 DC Amplifier	250 mv dc to 7 volts dc	Same
03, 04, 05, 06 DC Amplifier	From temperature sensors with resistances from 100 ohms to 23.5 ohms	Same
40, 41 DC Attenuator	+ 7 to + 50 vdc (40) -7 to -50 vdc (41)	Same
10 AD-DC Converter	0 to 5 vac rms and all intermediate ranges through 150 vac rms (400 to 3200 cps)	Same
30, 31 Phase Converter	0.5 to 5 vac rms (30) 5 to 50 vac rms (31 400 to 800 cps	Same
20 DC Low Gain Amplifier	± 2.5 vdc and all intermediate ranges to ± 35 vdc	Same

Signals from all rack sensor transducers are connected to a signal conditioner functionally similar to that used in the Apollo data subsystem. The signal conditioner normalizes the sensor signals to the standard 0-to-5 vdc full scale level acceptable to the data system analog-to-digital converter. The leads from the signal conditioner are connected to a subcommutator that switches its single output to each signal in turn, at a rate of one per second, synchronized with the Apollo data system commutator unit. At the output of the subcommutator, then, each sensor signal, properly normalized, is present in turn for one second.

The single subcommutator output lead is routed to the Apollo data subsystem main commutator, which will sample 200 signals per second, including the subcommutation lead, and enter each value ultimately in the telemetry data frames transmitted to the ground network. Connections between the external device and CM for this purpose are one signal lead from the subcommutator, and one synchronizing signal lead.

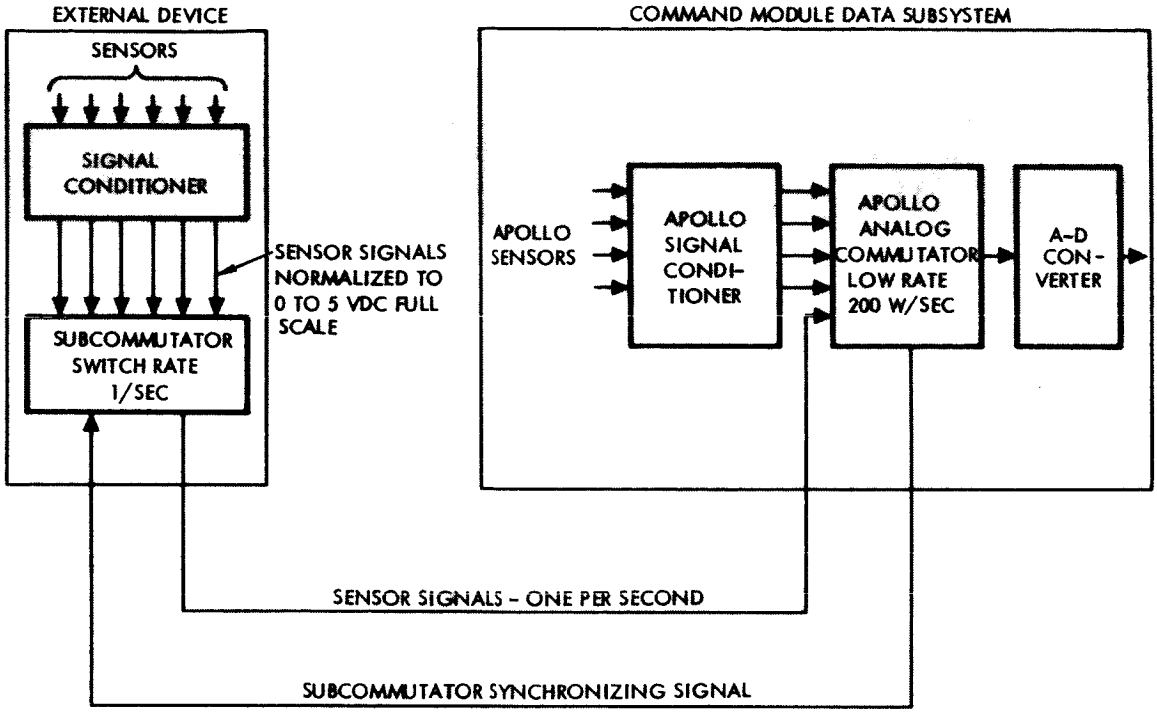


Figure 10. External Device Subsystem Telemetry Arrangement

System Operation

Experiment data-system operational characteristics, and the salient features of NASA experiment data recovery, are summarized in the following paragraphs.

Information flow, within the framework of experiment, spacecraft, ground network framework, is illustrated in Figure 11. The experiment data system shares time with the Apollo Unified S-band rf communications subsystem, the junction between Apollo and the experiment system being at the Apollo pre-modulation processor. For experiment data telemetry, the data are converted by various means into Apollo-format PCM frames and/or analog data, and transmitted in real time through the Apollo communications equipment, or, more commonly, stored during orbit for transmission by the same means during spacecraft-ground access periods. For experiment up-data link commands, a vehicle code different from that assigned the Apollo spacecraft will be provided for the experiment package, providing the same additional command-channel capability for the experiments as for the spacecraft.

As presently envisioned, the time constant for ground command reaction to experiment telemetry data will not need to be less than one an orbit period, (about 90 minutes). No tight closed-loop operations are planned for NASA

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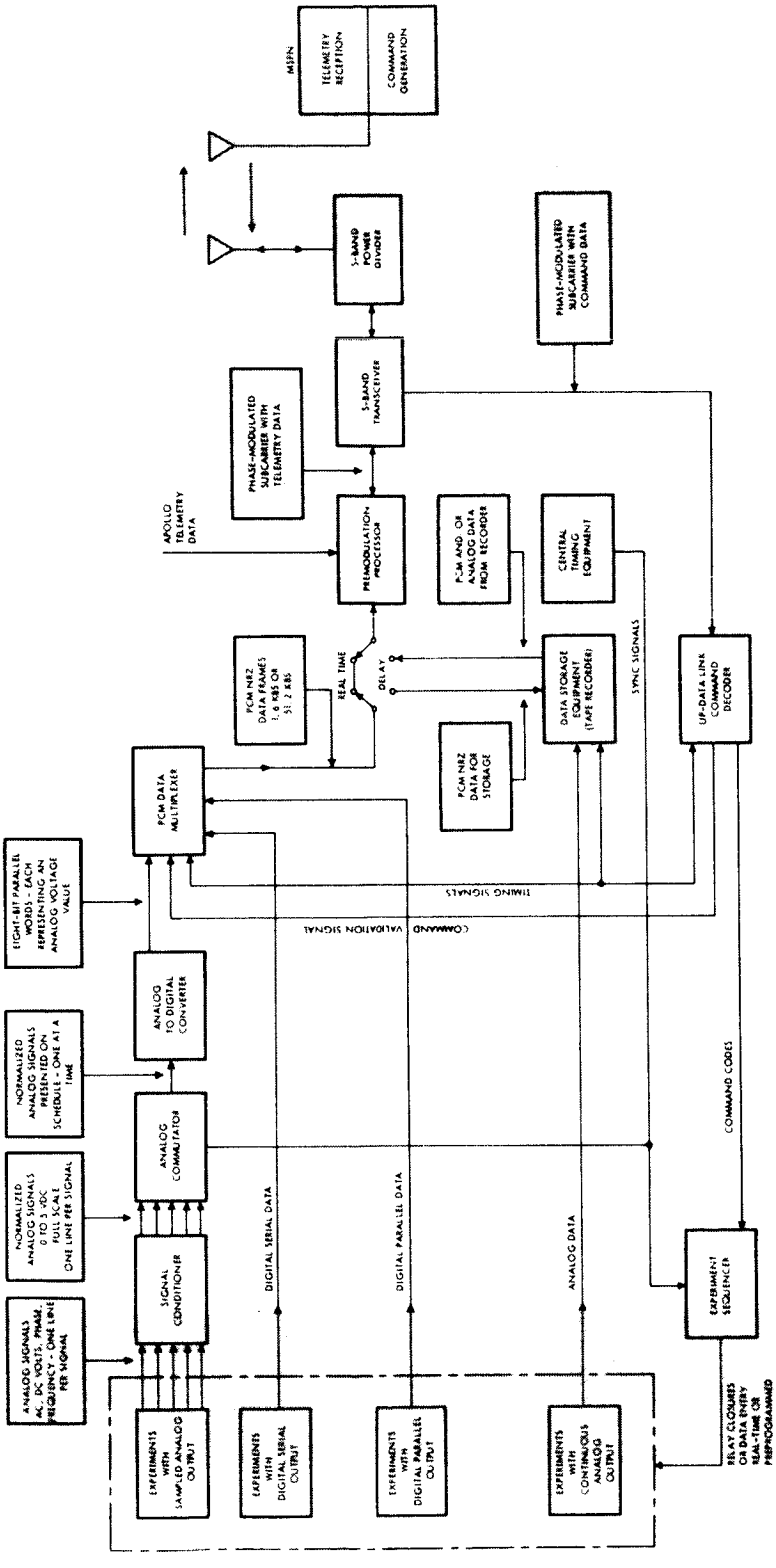
MSFN
TELEMETRY RECEPTION
COMMAND GENERATION
S-BAND POWER DIVIDER
S-BAND TRANSCEIVER
PHASE-MODULATED SUBCARRIER WITH COMMAND DATA
PHASE-MODULATED SUBCARRIER WITH TELEMETRY DATA
APOLLO TELEMETRY DATA
PREMODULATION PROCESSOR
PCM AND/OR ANALOG DATA FROM RECORDER
CENTRAL TIMING EQUIPMENT
PCM NRZ DATA FRAMES 1.6 KBS OR 51.2 KBS
REAL TIME
DELAY
DATA STORAGE EQUIPMENT (TAPE RECORDER)
SYNC SIGNALS
UP-DATA LINK COMMAND DECODER
PCM NRZ DATA FOR STORAGE
PCM DATA MULTIPLEXER
EIGHT-BIT PARALLEL WORDS - EACH REPRESENTING AN ANALOG VOLTAGE VALUE
TIMING SIGNALS
COMMAND VALIDATION SIGNAL
ANALOG TO DIGITAL CONVERTER
NORMALIZED ANALOG SIGNALS PRESENTED ON SCHEDULE - ONE AT A TIME
ANALOG COMMUTATOR
NORMALIZED ANALOG SIGNALS 0 TO 5 VDC FULL SCALE ONE LINE PER SIGNAL
SIGNAL CONDITIONER
ANALOG SIGNALS AC, DC VOLTS, PHASE, FREQUENCY - ONE LINE PER SIGNAL
DIGITAL SERIAL DATA
DIGITAL PARALLEL DATA
ANALOG DATA
COMMAND CODES
EXPERIMENT SEQUENCER
EXPERIMENTS WITH SAMPLED ANALOG OUTPUT
EXPERIMENTS WITH DIGITAL SERIAL OUTPUT
EXPERIMENTS WITH DIGITAL PARALLEL OUTPUT
EXPERIMENTS WITH CONTINUOUS ANALOG OUTPUT
RELAY CLOSURES OR DATA ENTRY REAL-TIME OR PREPROGRAMMED

Figure 11. NASA Experiment Data Communication Subsystem Information Flow Diagram

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experiments. Since the spacecraft rf equipment is to be time-shared, and available for experiment data transmission all the time except for four to seven minutes per orbit, and data rates, formats, and subcarriers are the same as for Apollo, ground network provisions for rf transmission/reception, subcarrier generation/reception and telemetry or command data handling may remain the same as those for the Apollo mission, with no equipment modifications. All experiment data-retrieval will be accomplished by one of three methods: (1) Real-time telemetry of data from experiments performed during spacecraft-ground communication access periods, permits limited high-rate ground recovery of data; (2) Data storage during orbit, with high-rate transmission during communication access, will be the most common mode of PCM and analog data recovery; (3) Some experiment data, typically film records and magnetic tape reels, may be physically retrieved from the experiment package by an astronaut for return to earth in the CM. There is no provision for data capsule ejection from the spacecraft for any NASA experiments.

Figure 11 illustrates information flow between components of the experiment data system. Three categories of experiment data (sampled analog, digital serial, and digital parallel) are sampled, digitized if necessary, and incorporated in Apollo data frame format. Where necessary for complete signal recovery, analog signals are stored for subsequent playback, or transmitted to the ground in real time. The premodulation processor serves as junction point between the experiment data acquisition subsystem and the down-link transmission loop. The premodulation processor accepts real-time or stored data for transmission, at a 51,200-bits per second rate for digital data, and up to 5 kilocycles in frequency for analog data. Up-data link commands for the experiment package are distinguished from the Apollo spacecraft commands by a different vehicle code acceptable only to the experiment up-link decoder. The decoder signals are routed to the experiment sequencer for relay contact implementation.

AIR FORCE MISSION SYSTEM

Data system design for Air Force experiment data acquisition, recovery, and command is similar in some respects to that adopted for NASA experiment data operations but is modified by different experiment requirements and procedures. On the basis of preliminary examination of experiment requirements and in the context of existing Air Force facilities and practices, the following data-system design ground rules were established:

1. Real-time and stored data transmission of experiment data are both required for the orbit altitudes and inclinations under consideration.

2. Modifications to existing Air Force ground-network facilities are to be minimized by adoption of a data system in use or under consideration by the Air Force.

3. Some of the data rates for Air Force experiments are very high and beyond the capacity of conventional telemetry system, but the bulk of experiment telemetry requirements (perhaps 90 to 95 percent) are modest. A standard telemetry system will be provided to satisfy all but the extreme requirements; these requirements will be handled on an individual, special-equipment basis as part of the experiment involved.

   The space-ground link subsystem (SGLS) vehicle and ground equipment recently developed for AFSC-SSD by Space Technology Laboratories is intended for application to all SSD manned and unmanned spaceflight programs, and was selected, with some modification, for this program.

While there is a gross similarity between the NASA system and SGLS, in capability at least, differences in rate and format preclude a simple interface between SGLS and the Apollo spacecraft and ground data subsystems. The experiment-data communication subsystem for Air Force experiments will be entirely separate from the spacecraft subsystems with no functional interface and no connection, except for power.

The SGLS equipment is designed to be preset to one of a variety of configurations in terms of data rates and sampling formats. Table 13 shows the data rate range; a low-rate and a high-rate range may be selected before launch for transmission by the SGLS transmitters.

For purposes of determining optimum system configuration, a preliminary evaluation of acquisition rates and periods for some of the AF experiments was made. In this investigation an attempt was made to identify the extreme requirements, which are listed in Table 14.

The data above represents extreme requirements, in terms of the earlier discussion, and provisions for acquisition and recovery of this data will be by means of special equipment to be designed and provided by the experiment designer. The remainder will be accommodated by the experiment data system.

The requirements for acquisition periods over seven minutes (the maximum spacecraft-ground network access period) dictate the addition to the SGLS of a data storage device to permit experiment data transmissions on a delayed and/or piecemeal basis, in other than real time.

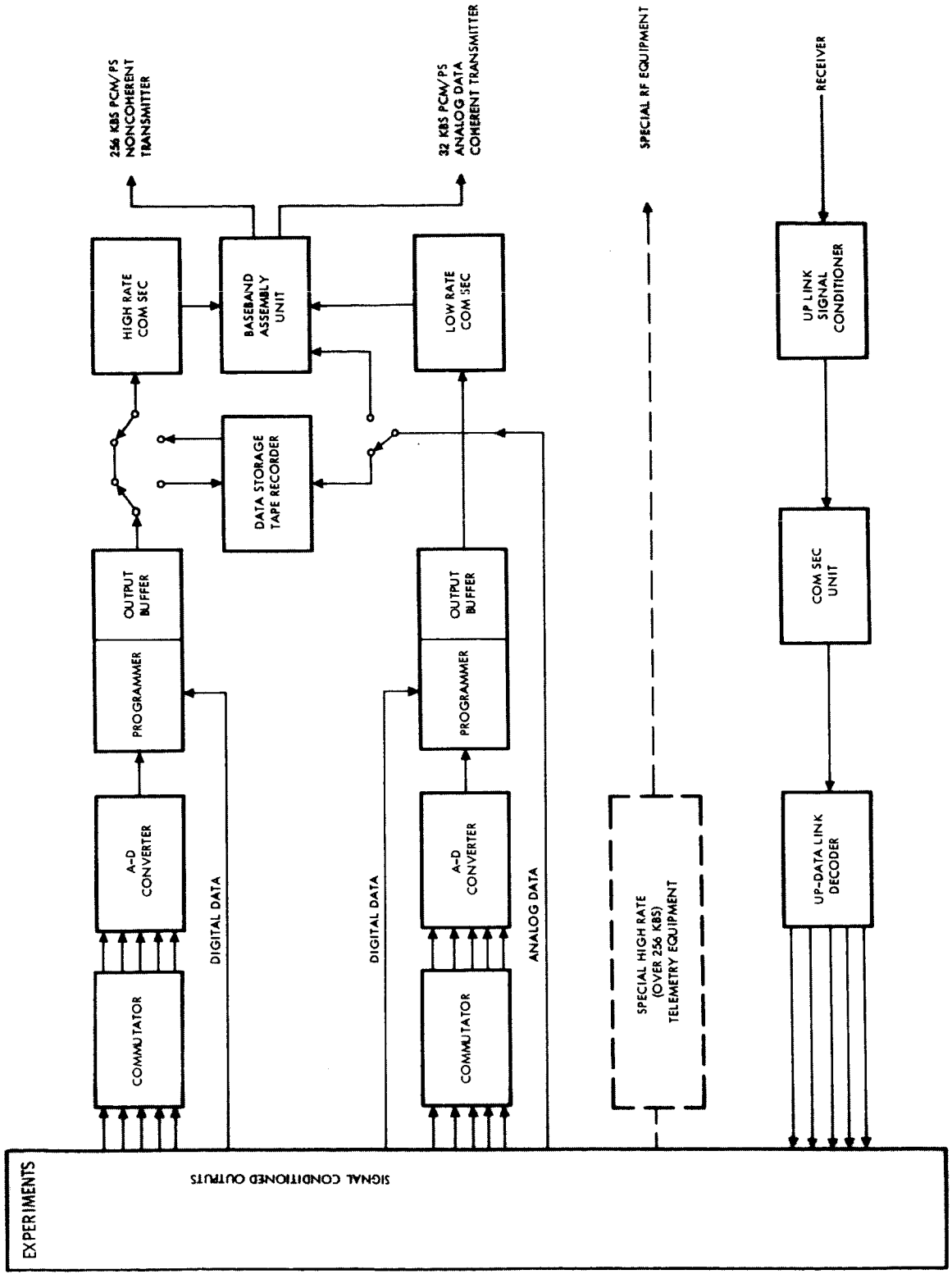

Figure 12. Air Force Experiment Data System Functional Block Diagram

Table 13. SGLS PCM Data Rates

Low Rates (Bits per second)	High Rates (Bits per second)
7.81	128, 000
31.25	256, 000
62.5	512, 000
250	768, 000
500	1, 024, 000
1, 000	
1, 600	
4, 000	
8, 000	
32, 000	
51, 200	
64, 000	

Table 14. Sampling of Air Force Experiment Data Acquisition Rates and Collection Periods

Bit Rate (Bits per second)	Period (min)
25, 000	15
900, 000	1.1
1, 600, 000	0.17
11, 000	30
62, 500	0.67
20, 000	10

These and other experiment and mission considerations dictate the data system design illustrated in Figure 12. The primary components of the system are the data handling portions of the SGLS and a separate and additional data storage tape recorder. Table 15 lists the capabilities of the system in terms of acquisition and transmission bit rates and times.

Table 15. AF Experiment Data System Characteristics

Mode	Acquisition (Record)			Transmission (Playback)		
	Rate (kbs)	Tape Speed (ips)	Time (min)	Rate (kbs)	Time Speed (ips)	Time (min)
1	256	40	4	256	40	4
2	32	5	32	256	40	4
3	4	5/8	256	256	40	4

The standard transmission rate will be 256 kbs, through the high-rate telemetry system and the noncoherent medium power SGLS transmitter. The three modes listed will accommodate all but the extreme requirements shown in Table 14 both in bit-rate and duration. This design permits greater independence, in experiment scheduling, of spacecraft position in reference to ground network remote sites than a strictly real time design. A brief functional description of system components follows:

Data storage equipment - This tape recorder will be similar in many respects to the Apollo DSE. Data will be impressed on one inch wide magnetic tape on eight tracks, with a ninth track provided for a clock signal. Tape length will be about 800-feet; bit density will be 800 bits per track per inch, with a total storage capacity of about 61.5×10^6 bits. Physical characteristics and power consumption are expected to be the same as for the Apollo DSE: 28.5 pounds, 6 x 9.5 x 22 inches, 33 watts.

SGLS data equipment - The components described here are functional divisions of the digital telemetry (DTU) unit of the SGLS, which is a single physical entity composed of a chassis base with plug-in modules. For this reason, the physical characteristics of these components will not be given here. To the extent possible, SGLS terminology and nomenclature will be accompanied by Apollo data system equivalents in the text.

The DTU processes analog, digital, or discrete input signals into a time-multiplexed PCM format. Analog signals are quantized to either 4- or 8-bit accuracy, depending upon the requirements of a particular mission. Digital words and discrete inputs bypass the A-D converter and are combined with its digital output, along with synchronization and identification words. Digital word lengths may be in integral multiples of 4 bits.

The format consists of a nominal 128-word main frame based upon 8-bit words for a total of 1024 bits per frame. Up to 256 input channels can be made available with shorter average word lengths. Total system capabilities will allow future expansion to a total of 512 main frame inputs to be sampled for a maximum of 2048 bits per frame. Up to eight of these main frame words may be subcommutated by 16, 32, 64, or 128 with a maximum of 1024 bits per submultiplexer (8 bits per word, 128 inputs). Up to four words may be used for frame synchronization utilizing a pseudorandom code. No word synchronization bits are required.

The output signal is buffered, serial NRZ data. The basic elements of the PCM telemeter are the clock, programmer, multiplexers, and A-D converter. The clock supplies timing pulses to the programmer. High-bit-rate missions use a crystal oscillator module, while lower-bit-rate missions use a tuning fork. A buffered clock output is supplied for synchronization of other vehicle equipment, if desired. Also, the DTU format may be synchronized by external clock signals.

The programmer supplies timing pulses to the A-D converter, selection signals to the multiplexer, and synchronization. Operation at either of two predetermined bit rates may be selected by command.

GUIDANCE AND CONTROL SYSTEMS

In accordance with basic Block II philosophy, this section considers the guidance and navigation, and stabilization and control systems as a single entity.

The guidance and navigation system discussion presents results of performance and reliability analyses for the NASA and Air Force flights. The stabilization and control discussion presents the control requirements, reports the reliability analyses, and the reaction control requirements.

GUIDANCE AND NAVIGATION SYSTEM

During this study, four distinct vehicle configurations were examined: Configurations 1, D', C, and D. Because of the early flight schedule of the Configuration 1 and D' vehicles, it was assumed that only the unmodified Apollo Block II G&N system would be available for use on these flights. For the later flights utilizing configuration C and D vehicles, it was assumed that modifications for life extension could be made to the Block II system or that a new G&N system could be used if necessary. Configurations C and D are treated together in the following discussions, because the variations in configuration do not significantly affect the G&N system.

APOLLO BLOCK II G&N SYSTEM

The Block II G&N system is divided into three major subsystems: inertial, optical, and computer. The inertial subsystem is used to provide a precise inertial attitude reference so that vehicle linear accelerations and spececraft attitude can be measured with respect to inertial space. The inertial subsystem is composed of the inertial measurement unit (IMU), portions of the power and servo assembly (PSA), and three electronic coupling units (ECU). The IMU is a three-gimbal stable platform which contains three stabilization gyros (25 IRIG Mod 2) and three accelerometers (16 PIPA). The inertial portion of the power and servo assembly contains the electronics and power supplies required by the IMU. The electronic coupling units are analog-to-digital and digital-to analog converters which transmit the three platform gimbal angles between the IMU and digital computer. The inertial ECU's can also provide analog attitude error signals to the S-IVB instrumentation unit as backup to the Saturn guidance system and attitude error information for display to the astronaut.

The astronaut uses the optical subsystem to make precision optical measurements for alignment or realignment of the IMU and for navigational data. Navigation sightings in earth orbit can be made by landmark tracking, star-to-horizon sightings, and star occulations. The optical subsystem consists of the scanning telescope, sextant with horizon photometer and star tracker, two ECU's, portions of the PSA, and associated displays and controls. Under Block II operating philosophy, the ground network is the prime source of navigational data.

The computer subsystem consists of the Apollo guidance computer and two display and keyboard units. The computer is the primary data-processing element of the G&N system. It contains the mechanization of all required guidance, navigation, and control functions, and issues discrete bit and control signals to other G&N system components and other space-craft systems. The computer has a random-access core-type memory of 36,864 words in fixed rope memory, and an erasable memory of 2,024 words (15 bits plus parity). The computer employs dual NOR-gate logic with 36 programmable instructions. The DSKY's are used for manual data insertion and program control, and also for program display. Block II G&N equipment weight is listed in Table 16.

APOLLO X G&N SYSTEM

During the course of the Apollo X study reported in SID 64-1860-20, several changes to the Block II G&N system were recommended to provide greater life extension. These changes include turning off the digital computer and the power for the inertial component (PIPA's and IRIG's) suspension when not required. Under normal Block II mechanization, the digital computer is operated in a standby mode during which it supplies a precise time-sync signal to the communications system. This function can be accomplished in the communications system by more reliable methods. Also, the component suspension power is supplied continuously under Block II mechanizations, although it is only required when the inertial subsystem is in use. The inertial component heaters, however, must operate continuously. Therefore, it was recommended that redundant heater power slip-rings be used to improve reliability of the IMU heating loop. These changes significantly improve reliability by minimizing the reliability degradation during nonoperating periods. For missions requiring long G&N operating periods, it would still be necessary to provide spares to attain high system reliabilities.

MISSION ANALYSES

The wide variety of missions that have been examined impose different requirements on the G&N system. In this section, the requirements imposed by various groups of missions during the boost, orbital, and reentry phases will be examined.

Table 16. Apollo Block II G&N Equipment Weight

Unit	Weight (lb)
ELECTRONICS	
IMU and coolant hoses	42.9
Navigation base	17.0
Power and servo assembly	58.2
Electronic coupling unit	36.0
Apollo guidance computer	69.0
Bellows assembly	12.7
Signal-conditioning assembly	3.9
OPTICS	
Sextant	18.7
Telescope	14.3
Optical base	17.0
Eyepieces	4.2
Optical shroud	3.1
DISPLAYS	
Displays and controls - navigation	12.1
Display and keyboard (2)	35.0
Map and data viewer	11.5
LOOSE-STORED EQUIPMENT	
Film cartridges	2.5
Eye relief eyepieces	1.5
Optics cover	1.6
CABLING	
Cabling MIT	38.8
G&N system Total	400.0

Boost Phase

Four distinct types of missions are scheduled to be flown. These are: (1) low inclination missions, (2) rendezvous missions, (3) high inclination missions, and (4) synchronous missions. For all missions, the spacecraft G&N system would be used in a monitor and backup mode during booster thrusting phases. Electrical interfaces have been provided so that under astronaut control, attitude error signals (roll, pitch, and yaw) would be supplied to the Saturn guidance system.

The low-inclination missions consist of Flights 209, 211, 215, 211, AF-1, and AF-2. For all of these flights, a Saturn-IB is used for boost into an 80-nautical-mile parking orbit. The spacecraft G&N system would provide the necessary guidance commands for thrust vector control of the SPS engine for orbit transfer and injection into the 200-nautical-mile orbit. The Apollo G&N can adequately meet the loose tolerances that are required for orbit circularization. Flights 215 and 221 differ from the others, because an empty S-IVB (for use as a counterweight in a rotational experiment) is still attached to the external device during SPS firing. This difference may necessitate a change to the cross product steering equations used for thrust vector control to reduce disturbances caused by the center of gravity displacement. The change of a single number in the steering equations however, will result in the application of thrust in a constant inertial direction rather than a time-varing direction.

Flights 219, 229, and 230 involve rendezvous with other flights. In general, rendezvous can be achieved using on-board rendezvous radar, ground tracking or optical measurements, or a combination of all of these methods. On the Apollo LOR and Gemini missions, a rendezvous radar on the maneuvering vehicle is used as the primary terminal sensor. For both missions, ground tracking stations supply the required orbital parameters. As backup on both missions, optical sightings can be made within 10 to 20 miles with the aid of blinking lights on the Agena and LEM. The implications of adding a rendezvous radar to the Apollo CSM were investigated during the Apollo Block II effort, and the radar has been deleted. The radar requires that an additional 14 pounds would be added to the G&N system weight to provide an electrical interface for range, range rate, and analog angle data. For these rendezvous flights, the LEM radar would be placed on the external device.

Flights 507, 513, and 518 would use a Saturn V to achieve a high inclination orbit. The G&N would be used to generate the steering signals during the SPS firing for orbit injection and in a monitor mode at other times. The Apollo G&N can meet the performance requirements of this mission.

A synchronous orbit will be achieved on Flights 509, 516, and 521. The Apollo G&N provides active steering only during the adjustment of the final orbit and it is used in a monitor mode at other times. The synchronous mission imposes the most stringent requirements on the achievement of a specified orbit. However, ground tracking data can be used to define corrections that are required to refine the orbit further.

Orbital Phase

The primary G&N requirement imposed by the various experiments is precise vehicle stabilization. On the Apollo LOR mission, G&N vehicle stabilization is used to control attitude to $\pm$ 0.5 degrees and attitude rate to 0.05 deg/sec. The most stringent requirements imposed by the experiments are control of vehicle attitude to $\pm$ 0.1 degrees and attitude rate to 0.01 deg/sec. The longest period during which precise vehicle stabilization must be maintained is approximately one earth orbit (about 1.5 hours). The vehicle stabilization requirements can be met by modifying the digital computer program and will not require major changes to the system.

The contributing factors to the G&N error in local vertical stabilization are the initial alignment of the IMU (60 arc-seconds), the gyro bias drift (1 meru) and the spacecraft position uncertainty in orbit. Since these error sources are independent and random, they were combined by root-sum-square technique to determine the overall standard deviation. Figure 13 shows the local vertical uncertainty versus time as a function of the orbit

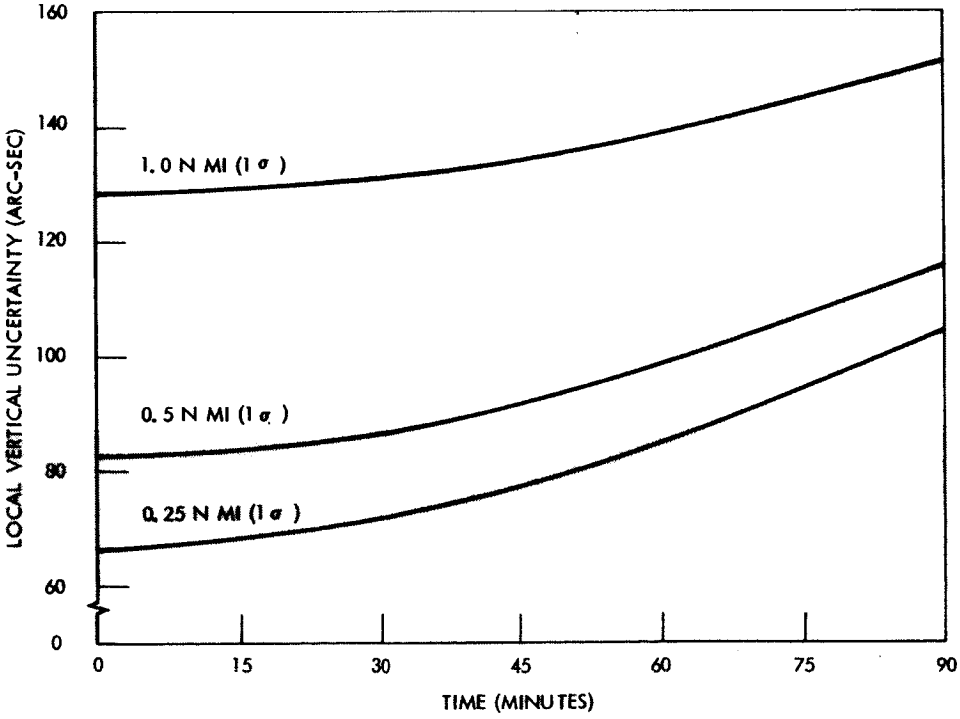

Figure 13. Local Vertical Uncertainty (1σ) Versus Orbit Position Accuracy

position accuracy. It can be seen that the attitude deadband requirement can be met with an allowance provided for vehicle control system uncertainty. It can also be seen that it will not be necessary to realign the IMU during an experiment run.

Because only the platform gimbal angles are measured directly, the vehicle attitude rate must be determined analytically. In the present computer program the attitude rate is based on the change in attitude during a 1-second sampling period. To meet the tighter requirements of the AES experiments, smoothing will be required to obtain the derived attitude rates.

The only AES experiments which require precise orbital maneuvering are the Syncom III Recapture and the Echo II Rendezvous. The only other vehicle translations would be performed manually as part of the behavioral experiments. Since the Syncom maneuver can be achieved by inducing a slow orbital drift, it is less stringent from a performance standpoint than the Echo experiment. Both experiments are to be flown on Configuration 1 vehicles thus limiting the G&N to the use of the Apollo Block II G&N system, which has no active terminal sensor.

The orbit transfer and coarse rendezvous with Echo must be accomplished based on ground tracking and visual sighting only. Due to the size of Echo (135-foot diameter), no difficulty in visually detecting the satellite is expected if the CM lighting conditions will permit the astronaut to detect at least a +2 magnitude star and a favorable Sun-Echo-CSM phase angle is available. Initially, visual sightings can be used to stablize the line-of-sight relative to the stellar background. At short distances, the known size of Echo can be used as a rough indication of relative range between the two objects as an aid in terminal maneuvering. Assuming that an object must subtend 12 arc-minutes [1] for shape to be detectable, the CSM must be within 6.4 nmi for relative range to be determinable.

A preliminary error analysis was performed to examine the accuracy with which transfer from an initial 200-nmi orbit to Echo II's altitude (assumed as 700 nmi) could be performed. The Echo II experiment is to be performed on the last day in orbit. It is assumed that any corrections which are required to reduce the initial out-of-plane error would be accomplished prior to the orbit transfer. The primary sources of error are due to the errors in knowledge of initial orbit parameters and due to errors in performing the transfer maneuver.

[1] Steedman, W. and Baker, C., Target Size and Visual Recognition, WADD TR-60-93, Feb. 1960

It is assumed that the parking orbit parameters are determined by ground tracking with the following 1 σ errors:

Error	Position (ft)	Velocity (fps)
Radial	1,000	0.2
Down range	3,000	0.75
Out-of-plane	3,000	0.75

A Hohmann transfer ΔV of 800 fps is required at perigee. The execution errors will be primarily due to the IMU misalignment uncertainty (0.6 mrad) and the accelerometer bias uncertainty (1.5×10^{-4}g). The resultant position errors will be negligible in comparison to the comparable initial position errors. The velocity errors are:

Error	Velocity (fps)
Radial	0.5
Down range	0.34
Out-of-plane	0.5

The errors from the two sources were then combined and used to form the intial co-variance matrix. A sensitivity coefficient matrix, which is a function of the transfer orbit eccentricity, was then used to propagate the errors from perigee to apogee. The resulting 1σ uncertainties at apogee are presented in Table 17.

Table 17. 1σ Uncertainties at Apogee

Error	Position (nmi)	Velocity (fps)
Radial	0.81	13.62
Down range	2.61	3.25
Out-of-plane	0.56	0.79

Since it is necessary to come within 1 to 2 nmi of Echo in order to obtain the desired camera resolution, some maneuvering would be required at apogee on a flyby mission. To provide additional time for taking pictures, it has been decided to circularize the orbit. The maneuver at apogee will be similar to that which was performed at perigee with comparable errors. Therefore, Table 17 is also indicative of the errors after circularization. The astronaut should be able to visually determine both the line of sight and relative range as aids in maneuvering closer to obtain the desired pictures.

Reentry Phase

For all of the AES missions, there are only two basic types of entries involved. These are entry from a 200-nmi orbit and from synchronous altitude. For Flight 513 which includes the Echo II experiment, the spacecraft returns to a 200-nmi orbit before performing the reentry maneuver. The Apollo Block II G&N is capable of performing a guided entry after a lunar mission with a 1σ G&N dispersion of 5.0 nmi at parachute deployment. This is adequate to meet all G&N entry requirements for the AES missions.

The various entries from 200-nmi orbit will differ from each other due to the effect of orbit inclination on the footprint orientation. However, the entry is less severe due to the reduced velocity as compared with a supercircular entry. Either the G&N or SCS can perform the entry maneuver.

The synchronous missions impose the most stringent entry requirements due to the similarity of entry conditions with those of the Apollo LOR mission. The first synchronous mission, Flight 509, is scheduled approximately 1 year after the first Apollo circumlunar flight, which will serve to verify the performance of the G&N for a guided entry.

MISSION/EXPERIMENT PROFILES

Due to the significant differences in operating requirements which are imposed by the AES missions as compared to those for the Apollo LOR mission, emphasis was placed on determing the reliability of the G&N system for the various mission profiles. Due to the limited number of alternate operating paths available within the G&N, a series reliability model. composed of the major G&N components was used. The component failure rates for both the Block II and Apollo X systems are given in Table 18.

To determine the system operating time, it was assumed that the G&N would operate continuously from launch until 1 hour after injection into the final orbit and from 1 hour prior to deboost until landing for all missions. Additional operating time in orbit would depend upon the experiment requirements. Due to the more severe reliability problem associated with the G&N as compared with the SCS, the integrated study approach was that all experiments that could be handled by the SCS were assigned to the SCS, and the G&N was assigned the remaining experiments. Table 19 presents a summary of the equipment operating time for all missions. The first row of flights all use either Configurations 1 and D' vehicles. The second row consists of the configurations C&D NASA flights and the third lists the C&D Air Force flights. An IMU alignment time of 15 minutes and a realignment time of 5 minutes was assumed for determing the optical subsystem (OSS)

time. The inertial subsystems (ISS) and computer subsystems (CSS) times are based on the experiment durations with additional time provided where necessary for alignment.

Table 18. Apollo G&N Failure Rates

Item	Mode	Failure Rate 10^6 hr	
		Block II	Apollo X
AGC	Operate	274.7	274.7
	Standby	37.6	0
IMU	Operate	132.0	130.0
	Standby	18.65	1.6
PSA	Inertial Operate	105.7	105.7
	Inertial Standby	9.5	0.2
	Optical	59.5	59.5
ECU	Inertial	217.9	217.9
	Optical	145.4	145.4
Optics		78.0	78.0
D&C		29.9	29.9

Flight 209

The G&N system is not required to support any experiment on this flight. The G&N is used only during the transit phases of the mission. The reliability of the Block II G&N for this mission is:

Subassembly	Probability of Success
AGC	0.9864
IMU	0.9933
PSA	0.9964
ECU	0.9990
OPTICS	0.99995
D&C	0.99997
Total G&N Reliability	0.9752

Table 19. G&N Operating Times

Flight and Subsystems	209		211		507		509		215		513	
Configuration and Phase	ISS/CSS	OSS	ISS/CSS	OSS	ISS/CSS	OSS	ISS/CSS	OSS	ISS/CSS	OSS	ISS/CSS	OSS
I & D' Vehicles												
Boost	1.4	0.2	1.4	0.2	4.9	0.3	6.0	0.5	1.4	0.2	4.9	0.3
Orbital	2.0	0.5	27.7	7.5	129.0	8.8	4.0	0.9	80.0	13.5	72.5	25.0
Entry	0.7	-	0.7	-	0.7	-	5.4	0.3	0.7	-	0.7	-
Total	4.1	0.7	29.8	7.7	134.6	9.1	15.4	1.7	82.1	13.7	78.1	25.3
C & D, NASA	219		221		518		229		230			
Boost	3.4	0.3	1.4	0.2	4.9	0.3	3.4	0.3	3.4	0.3		
Orbital	11.7	1.5	2.0	0.5	136.2	41.3	2.0	0.5	2.0	0.5		
Entry	0.7	-	0.7	-	0.7	-	0.7	-	0.7	-		
Total	15.8	1.8	4.1	0.7	141.8	41.6	6.1	0.8	6.1	0.8		
C & D, AF	AF-1		AF-2									
Boost	1.4	0.2	1.4	0.2								
Orbital	218.0	103.7	37.7	11.7								
Entry	0.7	-	0.7	-								
Total	220.1	103.9	39.8	11.9								

* Legend:

ISS - Inertial subsystems

CSS - Computer subsystems

OSS - Optical subsystems

Flight 211

The G&N system is used to provide precise vehicle stabilization for Experiment 0904 — LMS Checkout during the first seven days of this mission. The component reliability is shown below for each system component and in **Figure 14 as a function of time.**

Subassembly	Probability of Success
AGC	0.9665
IMU	0.9833
PSA	0.9899
ECU	0.9924
OPTICS	0.9994
D&C	0.9998
Total G&N Reliability	0.9328

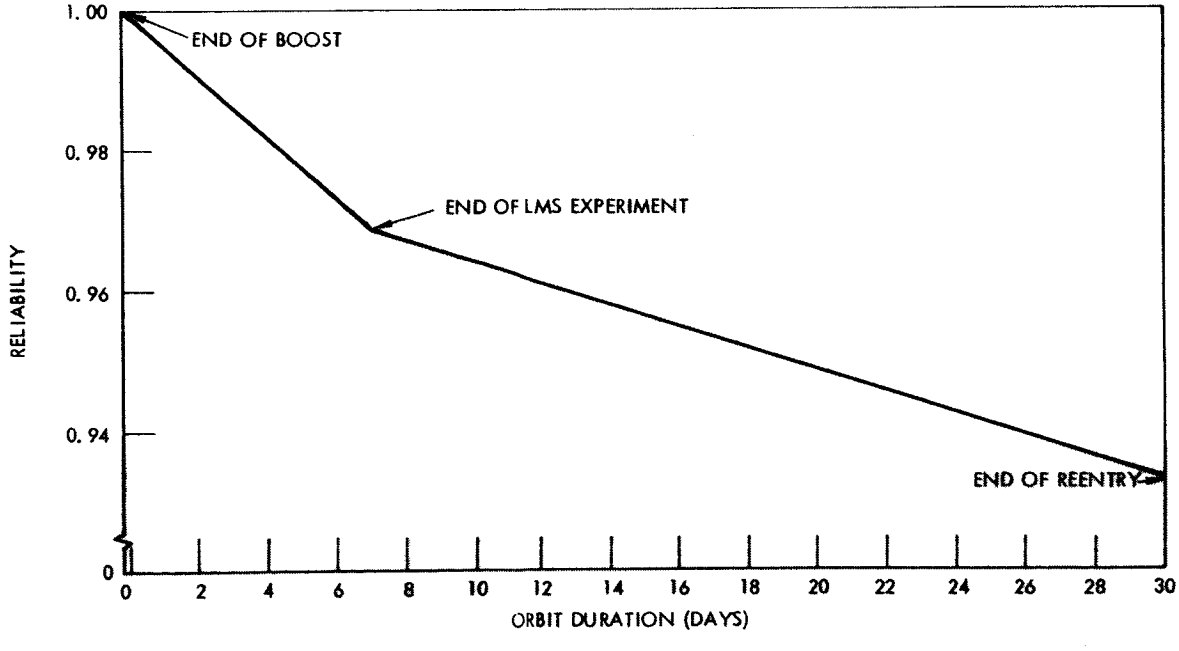

Figure 14. Apollo Block II G&N Reliability for Mission 211

Flight 509

At the end of this mission, the G&N system will be used during the combined Experiments 1603, Syncom III Recapture, and 1601, Orbital Maneuvering and Docking. Of all the early Block II missions, this will be the most critical mission from a reliability standpoint because of the desirability of having the G&N system operational during the supercircular reentry. The G&N reliability is presented below.

Subassembly	Probability of Success
AGC	0.9834
IMU	0.9918
PSA	0.9951
ECU	0.9964
OPTICS	0.99986
D&C	0.99995
Total G&N Reliability	0.9670

Flight 215

The G&N system will be used to provide precise vehicle stabilization for a continuation of Experiments 0901 and 0902, Earth Mapping, which were also flown on Flight 507. The system reliability for this flight is:

Subassembly	Probility of Success
AGC	0.9684
IMU	0.9845
PSA	0.9882
ECU	0.9803
OPTICS	0.9989
D&C Reliability	0.9996
Total G&N Reliability	0.9222

Flight 513

The G&N system is operated during Experiments 0802 A to D (each of which utilize an IR radiometer, IR spectrometer, microwave spectrometer, and a star tracker); Experiment 0502 Magnetic Fields, and Experiment 1602, Echo Rendezvous. Local vertical stabilization is provided for Experiments 0802 A and B and for 0502 since there is no local vertical mode in the Block II SCS. The microwave experiment, 0802 C, requires attitude stabilization of 0.15 degrees. The star tracker experiment is quite similar in operation

to that required to obtain star occulation measurements for navigation. Most of the experiments performed are done frequently (2 to 6 times each per day) and are of short duration (20 minutes or less). In order to reduce the optics operating time and the number of system standby to operate cycles, it has been assumed that the experiments could be grouped so that only 100 alignments would be required. This represents a decrease from the 203 alignments which would be required if each experiment run were separate. The probability of successfully completing the Echo experiment would be enhanced if the experiment were performed early in the mission instead of on the last day. The G&N reliability is shown below.

Subassembly	Probability of Success
AGC	0.9691
IMU	0.9849
PSA	0.9878
ECU	0.9795
OPTICS	0.9980
D&C	0.9992
Total G&N Reliability	0.9210

Flight 219

Flight 219 is the first flight on which a system other than the unmodified Block II G&N can be used. For the following flights the improvement in reliability which can be achieved by use of the Apollo X G&N will be shown. This flight is unique also in that it is the first flight during which a rendezvous and docking with another AES flight will be performed. During the orbital phase, the G&N system will be used in Experiment 1407 to evaluate the use of various types of measurements (landmark, stellar, horizon sensor and doppler-altimeter) for onboard navigation. The probability of success for Block II and Apollo X are shown on the following page.

Subassembly	Block II	Apollo X
AGC	0.9565	0.9957
IMU	0.9783	0.9962
PSA	0.9882	0.9980
ECU	0.9963	0.9963
OPTICS	0.99986	0.99986
D&C	0.99995	0.99995
Total G&N Reliability	0.9210	0.9861

As can be seen from the above data, significant improvements in reliability can be achieved by use of the previously described modification for the Apollo X system. To attain higher reliabilities, sparing would be required. Assuming that equipment sparing could be achieved at any time during the orbital phase and at any time in the parking orbit prior to the last hour before orbit transfer, a reliability of 0.9942 can be obtained by the addition of a spare ECU, AGC and PSA. This would add 163.2 pounds to the weight. It was assumed that box-level sparing rather than module-level sparing would be used due to hermetic sealing requirements.

Flight 221

The G&N is not required for any experiment support during this mission. This mission is unique in that the vehicle will be rotating at various rates during the entire 45-day mission duration. The system reliability is presented on the following page.

Subassembly	Block II	Apollo X
AGC	0.9592	0.9989
IMU	0.9796	0.9977
PSA	0.9893	0.9993
ECU	0.9990	0.9990
OPTICS	0.99995	0.99995
D&C	0.99997	0.99997
Total G&N Reliability	0.9286	0.9949

The Apollo X G&N can meet high reliability goals without sparing when it is only used during the transit phases on a 45-day mission.

Flight 518

During the orbital phases of this mission, the G&N provides vehicle attitude stabilization for the Earth-mapping experiments (0901, 0902, and 0903) and the microwave and star tracker experiments (0802 C&D). This mission requires the most operating hours of all the NASA experiment flights. The system reliability for this mission is:

Subassembly	Block II	Apollo X
AGC	0.9283	0.9618
IMU	0.9643	0.9802
PSA	0.9739	0.9825
ECU	0.9638	0.9638
OPTICS	0.9968	0.9968
D&C	0.9988	0.9988
Total G&N Reliability	0.8365	0.8887

To attain reasonable reliability levels for this flight it is necessary to provide spares for the CM G&N system or to provide a second system located in the rack for use as the prime system during the orbital phases. The reliabilities and additional weight for several alternatives are presented below:

Spares	Reliability	Weight (lb)
ECU, AGC, PSA, D&C	0.9740	175.3
ECU, AGC, IMU, PSA, D&C	0.9931	218.2
Apollo X G&N on Rack	0.9899	400.0

In determining the required spares, it was assumed that redundant optics could not be provided in the CM and it was desirable to limit the sparing of the IMU. An additional alternative would be to develop a new G&N system specifically to meet the long operating times and precise vehicle stabilization requirements of this mission.

Flights 229 and 230

From a G&N standpoint, these two missions are identical. Both missions require a rendezvous and docking phase. The G&N is not required to support any experiments during either flight. The G&N system reliability is presented below for both mission.

Subassembly	Block II	Apollo X
AGC	0.9587	0.9983
IMU	0.9793	0.9975
PSA	0.9891	0.9991
ECU	0.9986	0.9986
OPTICS	0.99994	0.99994
D&C	0.99997	0.99997
Total G&N Reliability	0.9272	0.9934

Flight AF-1

This is the first of the Air Force flights. It is the most severe flight for G&N reliability of all flights which were investigated during this study. The G&N is required to meet the vehicle stabilization requirements of: (1) 0.5 degree and 0.01 deg/sec for the visual tracking experiments (P-1, P-2, and P-3); (2) 0.3 deg for Experiment P-4, Electromagnetic Signal Detection; (3) 0.25 deg and 0.01 deg/sec for Experiment P-10, Multiband Spectral Observation and (4) 0.01 deg/sec for the ocean surveillance experiment (P-13). In addition to the high operating hours of this experiment, over 400 separate alignments and, consequently, system standby to operate cycles would be required due to the frequency with which the experiments are performed. Since most of the experiments involve the tracking of ground or space targets of opportunity, it is doubtful that the experiments could be physically scheduled in groups. The system reliability is as shown on the following page.

Subassembly	Block II	Apollo X
AGC	0.9114	0.9414
IMU	0.9559	0.9705
PSA	0.9631	0.9708
ECU	0.9389	0.9389
OPTICS	0.9919	0.9919
D&C	0.9969	0.9969
Total G&N Reliability	0.7790	0.8234

As in the case of Flight 518, several methods of sparing were investigated. The results are presented below.

Spares	Reliability	Weight (lb)
ECU, AGC, PSA, D&C	0.9576	175.3
ECU, AGC, IMU, PSA, D&C	0.9861	218.2
Apollo X G&N on Rack	0.9822	400.0

Flight AF-2

The G&N is used to provide: (1) precise vehicle stabilization of 0.5 deg and 0.01 deg/sec during the launch and retrieval of the RMU, Experiment P-7, and (2) local vertical stabilization for Experiments S-1 (VLF), S-2 (Mm Wave), and S-4 (Ionosphere Ducting). The G&N reliability is:

Subassembly	Block II	Apollo X
AGC	0.9511	0.9891
IMU	0.9756	0.9932
PSA	0.9853	0.9949
ECU	0.9896	0.9896
OPTICS	0.9991	0.9991
D&C	0.9996	0.9996
Total G&N Reliability	0.9037	0.9660

The reliability can be improved by sparing:

Spares	Reliability	Weight (lb)
ECU, AGC, PSA	0.9905	163.2
ECU, AGC, IMU, PSA	0.9970	206.1

STABILIZATION AND CONTROL SYSTEM

Analyses were conducted to evaluate the use of the Block II system to satisfy the control requirements associated with a series of mission profiles. Several vehicle configurations were considered. Control requirements were specified using experiment definition data and the results of mission analyses. Modes of operation and the associated durations were determined, reaction control system requirements were estimated, and experiment program modifications were investigated to alleviate tankage problems. Required modifications to the Block II control system were noted. Preliminary reliability analyses were conducted to determine redundancy or spare requirements. In general, it was concluded that a modified Block II system, in conjunction with the guidance and navigation equipment, can meet the requirements of the missions under consideration. The modifications are component sparing and minor system modifications.

The data resulting from the study are presented in a manner designed to facilitate its future use and each of the analysis areas, including methods, assumptions, and background material, is discussed. The appendix documents the results of the studies and is organized by mission. Fixed format tabular presentations aid in the location and comparison of data as well as the investigation of effects of mission profile variations.

CONTROL REQUIREMENTS

The control requirements analysis was conducted to determine attitude and drift rate tolerances, angular acceleration limitations, rotational maneuver characteristics, and modes of operation. In general, the study was based on the data contained in the individual experiment definition forms and the experiment scheduling input data (Volume II).

Attitude Hold and Rate Stabilization

The attitude hold accuracy requirements were extracted from the individual experiment definition forms. In cases where the experiment write-up indicated a need for attitude hold without specifying an accuracy, an estimate was made. The estimate was based on available experiment information and requirements associated with similar functions in other experiments. The attitude hold and rate stabilization requirements associated with the experimental phases of the various missions were tabulated by individual experiment. The data are contained in the individual data packages in the appendix.

Rotational Maneuvers

Many of the experiments, particularly the remote sensing of the earth's atmosphere, have directional requirements. It is, therefore, necessary to establish a particular vehicle orientation each time the experiment is conducted. Some experiments will, also, require maneuvers during the course of the experiment. These requirements are contained in experiments involving targets of interest or operations in the vicinity of, but external to, the spacecraft. In the latter case, maneuvers could be required to keep an external object within the field of view of the spacecraft windows.

Maneuver requirements were not specified in the experiment definition forms. The data shown in the individual mission packages represents preliminary S&ID estimates of the maneuver requirements associated with each experiment. The estimates are based upon available experiment data, experiment duration, and the total number of times the experiment is conducted. The experiment duration and frequency data used were preliminary; therefore, the maneuver requirements should be revised when better information is available.

Experiment scheduling was neglected in the analysis of maneuver requirements due to a lack of the necessary information. However, when scheduling data is available, it should be used to modify the estimated maneuver requirements. For example, if two experiments are run sequentially, and if both require the same vehicle orentiation, then a reorientation maneuver is not necessary at the beginning of the second experiment. Thus, as a result of scheduling considerations, the number of maneuvers per flight can be expected to vary from the data presented in this section.

Slewing rates of 0.5 degrees per second were assumed necessary for the on-orbit maneuvers; some maneuvers will be performed at lower slewing rates, particularly in cases where small magnitude reorientations are required. On the other hand, excessive time is required to perform large angular reorientations at low slewing rates. For example, approximately 15 minutes are required to perform a 180-degree maneuver at a slewing rate of 0.2 degrees per second. Under these conditions, the system operating time could become prohibitive, resulting in an extreme reliability degradation; therefore, a conservation decision was made to assume all maneuvers would be performed at a slewing rate of 0.5 degrees per second.

Modes of Operation

The control requirements associated with each experiment were analyzed to select a mode of operation for use during that experiment. The modes were selected to satisfy both functional and accuracy requirements and, also, to alleviate reliability problems. Study ground rules also constrained the selection of operating modes. On the early flights, for example, study ground rules dictated the use of Block II systems and it was necessary to satisfy all local vertical control requirements by a G&N prime mode of operation. In some missions, this resulted in severe reliability degradations. On the later flights, where subsystem modifications were allowed, an SCS local vertical mode was used to alleviate the G&N reliability problem.

The SCS components which are used in each mode of operation are shown in matrix form in Table 20. In addition, the cumulative time spent in each mode has been tabulated for the various missions. This data is presented in the mission packages section of the appendix. Reliability analyses and total electrical energy estimates were based on the above-mentioned data.

The SCS rotation mode presents particular problems which should be noted. This control mode is used during the rotational phase of Experiments 0101 and 0102. The vehicle is torqued about either the pitch or yaw axis to accelerate the nine rpm in steps of three rpm each. The vehicle is then decelerated in a similar fashion. A rotational profile is shown in Figure 15. Manual control is recommended for this operation. From a hardware standpoint, a pilot-controlled mode represents the least complex approach to the problem. Provision for an automatic rotational mode would require system modification. The crew time requirement represents the only known penalty associated with a manual rotational mode. As the total time per mission spent in this mode is small, however, the crew time penalty is almost negligible.

It is recommended that a pilot-in-the-loop simulation study be conducted to demonstrate the feasibility of the manual rotation mode. Some of the important parameter variations which should be investigated during the simulation include reaction-jet-thrust misalignments, thrustor failures, mass distributions, and sensor performance.

Some hardware modification problems are associated with the rotational phase of Experiments 0101 and 0102. These experiments require pitch and yaw rotation rates of 9 rpm or 54 deg/sec. The rate sensors used in the vehicle control system are capable or sensing rates up to 50 deg/sec. Thus if it is desired to use the standard SCS sensors, the rotational rate will have to be restricted to approximately 8.3 rpm. A problem also exists in that the maximum pitch or yaw rates that can be displayed on the flight

Table 20. SCS Components Used in Various Modes of Operation

Mode of Operation	Solenoid Driver Assembly	Control Electronics Assembly	Display Electronics Assembly	Gyro Display Coupler	Servo Amplifier (Thrust Vector Control)	Gyro Assembly	Flight Director Attitude Indicator	Gimbal Position and Fuel Pressure Indicator	Attitude Set Control Panel	Translation Control	Rotation Control	Horizon Scanners	Entry Monitor System
1. SCS Attitude Hold With Manual Maneuver	X	X	X	X		X	X		X		X		
2. SCS Rate Control (Minimum Impulse)	X	X				X							
3. SCS Rotation	X		X			X	X	X		X	X		
4. SCS Thrust Vector Control	X	X	X	X	X	X	X		X				
5. SCS Attitude Hold (Minimum Impulse)	X	X				X							
6. G&N Prime	X												
7. G&N Prime With Manual Maneuver	X		X			X	X				X		
8. SCS Local Vertical With Manual Maneuver	X	X	X	X		X	X		X		X	X	
9. Sun Orient	X					X						X	
10. Drift	X										X		
11. Ascent and Entry	X	X	X	X	X	X	X	X	X	X	X		X

director attitude indicator are 10 deg/sec. Thus, if it is desired to use the standard SCS display, a scaling modification will be required.

RELIABILITY ANALYSIS

Preliminary reliability studies were conducted for each of the missions. The results of these studies indicated that reliability goals of 0.98 could be met in most cases without sparing. Flights 221, 518, and 229 required minor component sparing to meet a 0.98 goal. Figure 16 illustrates the results of the analysis. The base system reliability is shown for each mission together with the incremental reliability gains achieved by sparing.

REACTION CONTROL REQUIREMENTS

Associated with stabilization and control system operating modes and time lines is the determination of the requirements imposed on the reaction control system by the experimental programs. Utilizing the system operating times and vehicle maneuver summaries previously generated, each flight was analyzed from the standpoint of reaction-control-propellant quantity and reaction-jet duty cycle. Since a study constraint was the use of available Apollo components, it was necessary to determine which flights (1) resulted in reaction control quantities in excess of those which could be supplied by the allowable tankage, and (2) placed requirements on reaction jet life and starts in excess of the present specifications.

System Summary

The reaction control system utilized on extended mission vehicles is the basic system used on the Block II Apollo vehicle with the exception that LEM reaction propellant tanks can be substituted for the present propellant tanks on certain flights (i.e., C and D configurations). With this substitution, it is possible to increase the propellant capability from the Block II value of 790 pounds to a maximum of 2280 pounds, using two sets of LEM tanks per quad. Table 21 lists the tankage combinations which are possible using the Block II and LEM tanks.

Table 21. Reaction Propellant Tankage Combinations

Tankage (fuel and oxidizer)	Usable Propellant (pounds)
4 SM Tanks	790
8 SM Tanks	1580
4 LEM Tanks	1140
8 LEM Tanks	2280

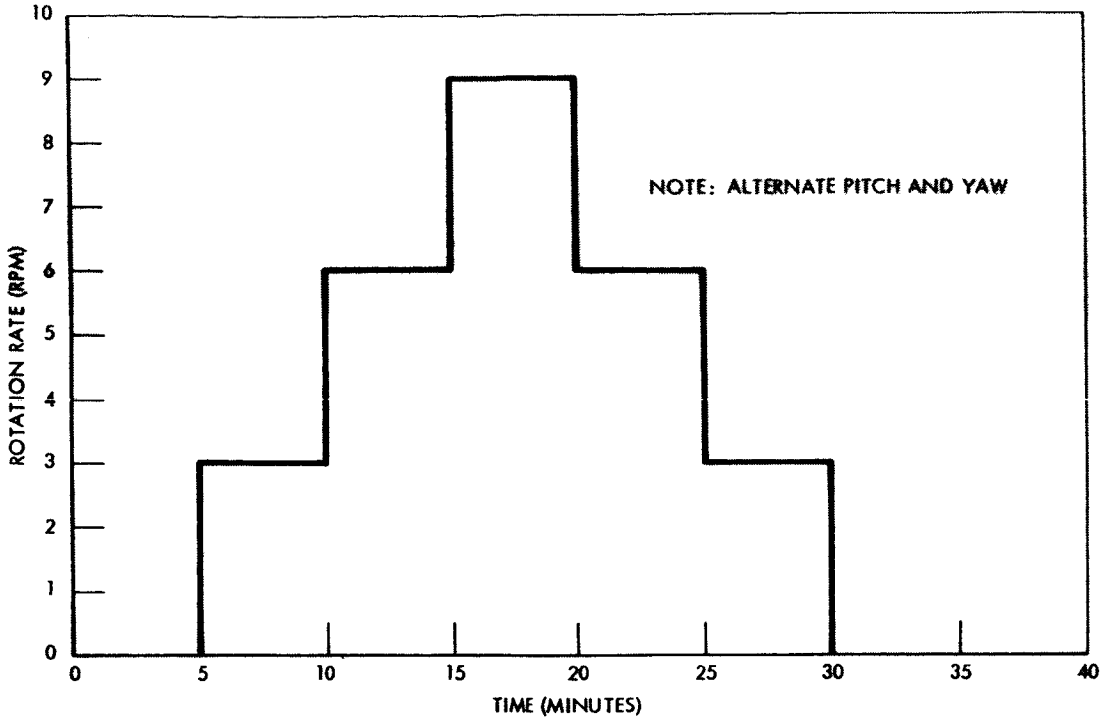

Figure 15. Rotation Profile for Experiments 0101 and 0102

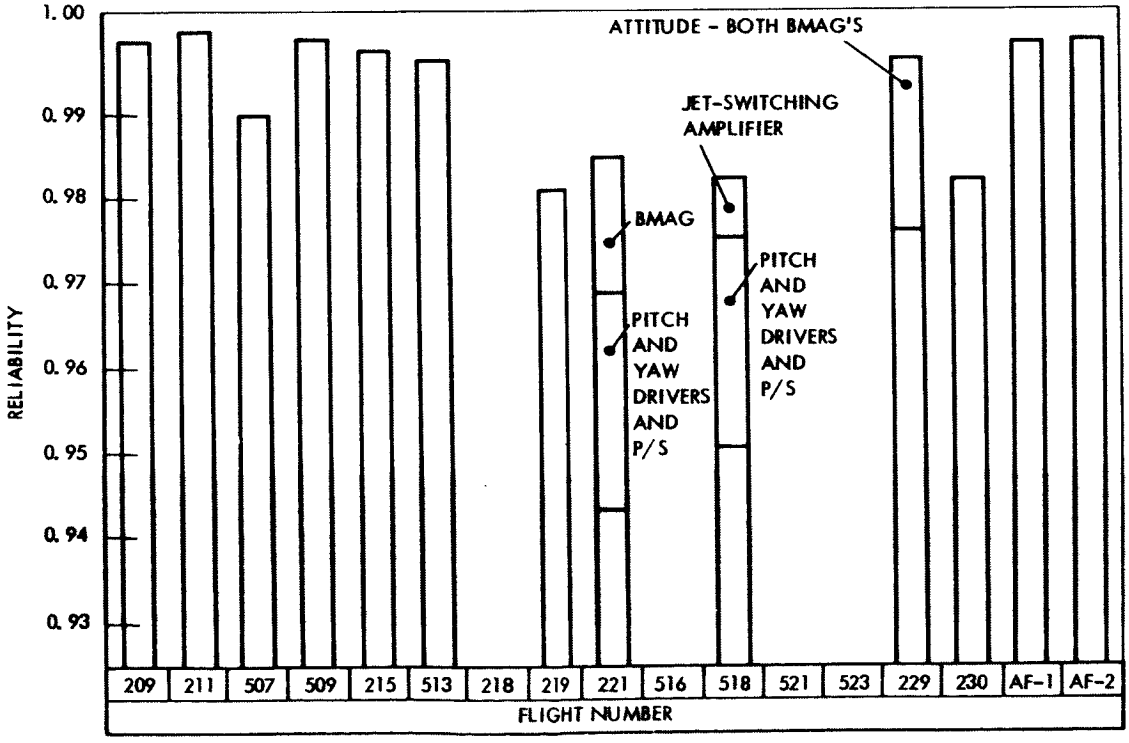

Figure 16. SCS Reliability for Various Missions

Attitude hold about each axis is controlled by four jets, two supplying positive control torques and two supplying negative torques. During limit cycle operation, each 100-pound-thrust jet fires with 0.6 pound-seconds of impulse. Vehicle rotational maneuvers are accomplished by two jets firing in couple to accelerate the vehicle to the desired rate and then two jets firing to reduce the rate at the conclusion of the attitude change.

Mission Analysis

For the analysis, each flight was separated into three phases; ascent, on-orbit, and preentry. These are the phases of flight during which vehicle stabilization and control is maintained by the reaction control system, using manually or automatically generated signals to effect vehicle translation, rotation, and attitude hold.

Control requirements during the ascent and preentry phases are essentially mission-independent, in that certain maneuvers are common to these phases in all missions. The on-orbit control requirements are highly mission dependent and result directly from the experiment program associated with each flight.

Ascent

Reaction control during the ascent phase is required to provide transposition and docking of the command and service modules to the rack in the manner of Apollo-LEM transposition and docking during the LOR mission. The entire configuration is then separated from the SIV-B boost stage and reoriented for orbit injection. The translational and rotational velocity requirements for these maneuvers are based on values being used in current Apollo studies. It is assumed that during ascent to orbit, four ΔV maneuvers are performed. Since the main propulsion tanks are not full for most flights, it is necessary to perform an ullage maneuver prior to each ΔV to insure propellant settling. Finally, a three-axis reorientation is performed to align the inertial platform.

The propellant quantities and jet duty cycles for the respective rotational maneuvers are based on man-in-the-loop simulation studies being conducted using the Apollo control system. Three-axis reorientations are assumed to be performed sequentially about the vehicle roll, pitch or yaw, and roll axes, since this type of a maneuver requires considerably less propellant (one-third less) than a manual maneuver performed about all three axes simultaneously. Table 22 is a summary of the vehicle maneuvers performed during the ascent phase.

Table 22. Ascent Phase Maneuver Summary

Maneuver	Comments
1. Separation from SIV-B and docking	ΔV = 4 feet per second
2. Transposition	ω = 5 degrees per second
3. Separation from SIV-B	ΔV = 3 feet per second
4. Reorientations (4)	ω = 0.5 degrees per second
5. Ullage maneuvers (4)*	at 9 seconds burn time each
6. Platform alignment	ω = 0.5 degrees per second
*Flight 507-2 Flight 509-1 Flight 513-2	

On-Orbit

Reaction control during the on-orbit phase is required for attitude-hold functions, vehicle maneuvering, and vehicle pointing associated with the experiments included in each flight. Propellant utilization rates for inertial attitude hold and local vertical attitude hold (longitudinal vehicle axis horizontal to the earth and in the orbital plane) were determined for each vehicle configuration. The propellant rates for attitude-hold were evaluated by determining the theoretical limit-cycle propellant and then analyzing the influence of gravity gradient and aerodynamic disturbance torques on the vehicle during limit cycle operation. These torques resulted in biased limit cycle operation, which for most flight configurations caused an increase in reaction propellant consumption. The propellant required for the maneuvers associated with each experiment was determined using the pilot-in-the-loop simulation data previously noted. Then, using the operating mode and rotational maneuver summary sheets, reaction propellant quantity and jet duty cycles were determined for each flight.

The reaction control system also supplies the velocity increments necessary to prevent orbit decay. Although altitude loss should not be critical during the 14-day missions, it could become so in the 45-day flights.

In the interest of consistency, the necessary propellant was included in each flight. The following equation is the expression which relates the daily velocity increment to vehicle weight, flat plate area, and drag characteristics:

$$\Delta V = \frac{18.5}{W/C_D A} \text{ feet per second per day}$$

Flights 219, 229, and 230 rendezvous with flights already in orbit, and flights 507 and 513 must perform rendezvous as part of their experimental programs. For these flights, it is necessary to include additional reaction control propellant to provide active vehicle control during the terminal rendezvous and docking phase. A ΔV capability of 30 feet per second has been included in the reaction propellant quantities of these flights.

Preentry

The preentry phase of the mission is that portion of the flight following the completion of the experimental program and prior to command module entry. It is assumed to consist of the last few mission orbits and the time up to CM-SM separation. The reaction control system provides local vertical attitude hold capability during the last three orbits of the flight. The inertial platform is aligned prior to deorbit and the external device is jettisoned. Finally, the service module is separated from the command module. Table 23 is a summary of the preentry maneuver requirements.

Table 23. Preentry Phase Maneuver Summary

Maneuver	Comments
1. Local vertical hold	≃ 3 hours
2. Platform alignment	ω = 0.5 degrees per second
3. External device separation orientation	ω = 0.5 degrees per second
4. External device separation	ΔV = 1 foot per second
5. Deorbit orientation	ω = 0.5 degrees per second
6. CM-SM separation orientation	ω = 0.5 degrees per second
7. CM-SM spearation	ΔV = 10 feet per second

Experiment Modifications

A preliminary evaluation of reaction propellant requirements for each flight and configuration indicated that, in a majority of the flights, these requirements exceeded the allowable propellant capacity. This capacity was limited to Block II tankage (790 pounds) on Configuration 1 vehicles, and two sets of LEM reaction propellant tanks (2280 pounds) on C and D configurations. Two facts were obvious from a consideration of the propellant breakdown for each flight: First, the attitude-hold propellant quantity for most flights is a small fraction of the total requirement (<10 percent). One exception to this is Flight 518 when a large amount of time is required for solar panel control and experiments using small attitude deadbands. Secondly, the rotational profiles associated with Experiments 0101 and 0102 require extremely high quantities of reaction propellant (80 to 130 pounds per spin profile). From these considerations, it appeared possible to effect a considerable propellant savings by reducing the number of scheduled spin profiles while leaving unaffected the remaining experiment program. Since these spin profiles are scheduled on all flights, with some flights requiring as many as 19 operations, a reduction in this number can be made without seriously compromising the overall experimental program. These reductions, together with a restriction on the targets of opportunity which can be viewed during each flight result in propellant weights for the respective configurations which are within the allowable capacity. Table 24 contains a summary of the above modifications as they affect each flight.

In addition to the experiment modifications mentioned above, there are several specific changes on individual flights. One example is the limitation on the artificial-gravity spin test on Flight 215. The restriction to using Block II reaction propellant tankage caused a reduction in the available acceleration from the planned 1 g to 0.18 g. This modification is explained later. Propellant distribution problems on Flight 518 resulted in a limitation on the number of times experiment 0802B (Test of an Infrared Scanning Spectrometer) could be performed. As originally scheduled, it was to be performed six times each day for a total of 270 times during the mission. However, because of quad distribution problems, it was necessary to limit the total number of tests to 213 for Flight 518-C and 154 times for Flight 518-D.

SPECIAL CONSIDERATIONS

Specific problems were encountered in Flights 215 and 221 because of the artificial-gravity rotational experiment, on Flight 518 because of the solar panels, and on Flight 507 because of guidance system reliability considerations. These problems are discussed in the following paragraphs, together with proposed solutions.

Table 24. Reaction Propellant-Experiment Modifications

Flight No.	Experiments 0101 and 0102 Rotational Sequence		Targets of Opportunity	Notes
	Scheduled	Proposed		
209	6	6	-	No changes necessary
211	12	5	84	
507	6	4	40	
509	4	3	23	
215	6	0	10	Artificial gravity test - 0.18 g at 50 feet
513	6	0	196	
219C	19	15	80	
219D	19	12	80	
221C	-	-	-	No changes necessary
221D	-	-	-	No changes necessary
518C	19	0	-	Run 0802B 213 times (scheduled as 270)
518D	19	0	-	Run 0802B 154 times (scheduled as 270)
229C	19	15	24	
229D	19	12	24	
230C	19	18	13	
230D	19	12	13	
AF-1	-	-	-	No changes necessary
AF-2	-	-	-	No changes necessary

Flight 507

The control requirements associated with Experiment 1201 (Local Vertical Within ± 35 Degrees for 96 Hours) represent a reliability problem for the G&N equipment. It is recommended that an SCS modification be investigated as a possible means of solving the problem. The roll and yaw control could be provided using an SCS attitude-hold type operation. The pitch control requirement could be satisfied using orbit rate torquing. This type of control would be adequate considering the 35-degree tolerance available. It is recommended that the hardware modification required to provide this mode of operation be defined in order to enable further evaluation of the problem.

Flight 215

One experiment on Flight 215 (0301) is a test of the deployment mechanism and spinup procedure to be used in the artificial gravity experiment that will be conducted for 45 days on Flight 221. On Flight 215, it was planned that the S-IVB counterweight be deployed and the configuration spun to produce the 1 g level that the vehicle will be subjected to on Flight 221. It was originally intended to perform this test at the end of the 14-day mission. If the S-IVB remains attached to the rack/CSM in the undeployed configuration, however, manual maneuvers require approximately 10 times more propellant than the rack/CSM alone because of the large vehicle moments of inertia. If the deployment test is performed at the end of the mission, approximately 5000 pounds of reaction propellant will be required for the experiment program. This is clearly in excess of the 790-pound capacity of the Block II reaction propellant tanks. For this reason, it is recommended that the deployment test and spinup be conducted as early in the mission as possible and preceding any other experiment work. The S-IVB could then be separated and the problem of maneuvering during the mission while attached to the S-IVB would be eliminated.

A full spinup to 1 g and despin requires 789 pounds of reaction propellant. This is essentially the capacity of the Block II tanks (i. e., 790 pounds) and does not include the ascent and descent requirements or the experiment propellant requirements. If the 0101-0102 spin profiles are eliminated from the experiment program but the remainder of the experiments performed, there are 270 pounds of reaction propellant available for a partial test of the deployment device. With this propellant weight, it is possible to spin the configuration S-IVB and rack/CSM to an angular rate of 20 degrees per second or 0. 18 g at the fully extended position and despin it.

The recommendations regarding experiment and procedural modifications for Flight 215 are:

1. Perform the artificial g deployment and spin test (0301) the first day of the mission and then separate the S-IVB.

2. Spin the configuration to provide an acceleration level of 0.18 g.

3. Eliminate the six spin profiles associated with Experiments 0101 and 0102.

Adoption of these recommendations will permit the mission to be performed using the Block II tanks.

Flight 221

The artificial g experiment (0301) requires a large amount of reaction control propellant, with the exact amount depending on the manner in which the experiment is performed. For this experiment, the S-IVB boost stage is attached to the rack/CSM with extendable tubes. With the empty S-IVB acting as a counterweight, the configuration is spun to produce an artificial gravity level during the flight. The purpose of the experiment is to study the effects of both spin radius and g level on the astronaut-subjects. Table 25 contains the selected sequence of spin radius and spin rate combinations which provide ample experimental variation while minimizing reaction propellant consumption. Propellant is a minimum for the selected sequence because the angular momentum value during each portion of the experiment is greater than the preceding phase (Table 25). It is therefore necessary to provide only positive (negative) increments of angular momentum during spinup (spindown).

This sequence is to be performed twice during the mission. The configuration is spun up and despun once about the pitch axis and once about the yaw axis. For a pitch maneuver, the vehicle would be accelerated by firing jets A_1 and C_3 (Figure 17). Because of the large moments-of-inertia of the deployed configuration, however, it requires 1230 pounds of reaction propellant to perform the sequence of radii and spin rates. If, however, the (+Z) translation jets (B_4 and D_2) are used, the jet arm is increased from 13.4 feet to as much as 52 feet during the sequential spinup (Figure 18.) Using this technique, the propellant required to spinup the configuration according to the selected schedule is 395 pounds. To despin the vehicle, the (-Z) translation jets (B_2 and D_4) are utilized. The same sequence of spin and despin is performed about the yaw axis using the positive and negative (Y) translation jets (A_4 and C_2, A_2 and C_4). Performing the sequence once about the pitch axis and once about the yaw axis ensures that the required propellant (1580 pounds) is evenly distributed among the four reaction propellant quads.

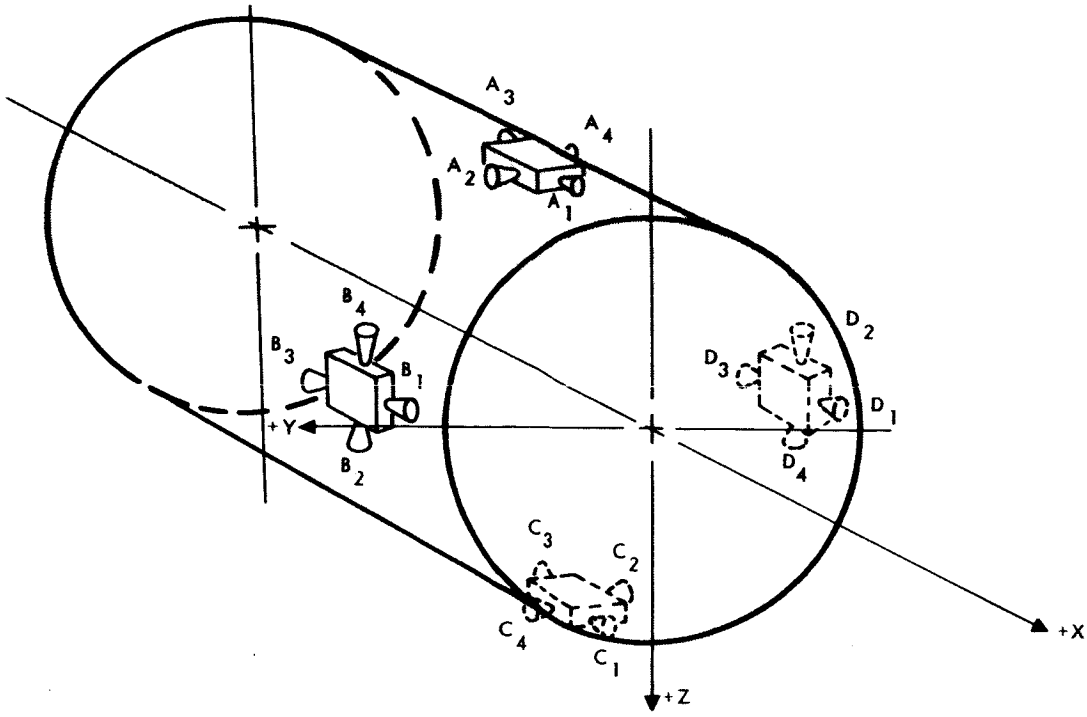

Figure 17. Reaction Jet Location Diagram

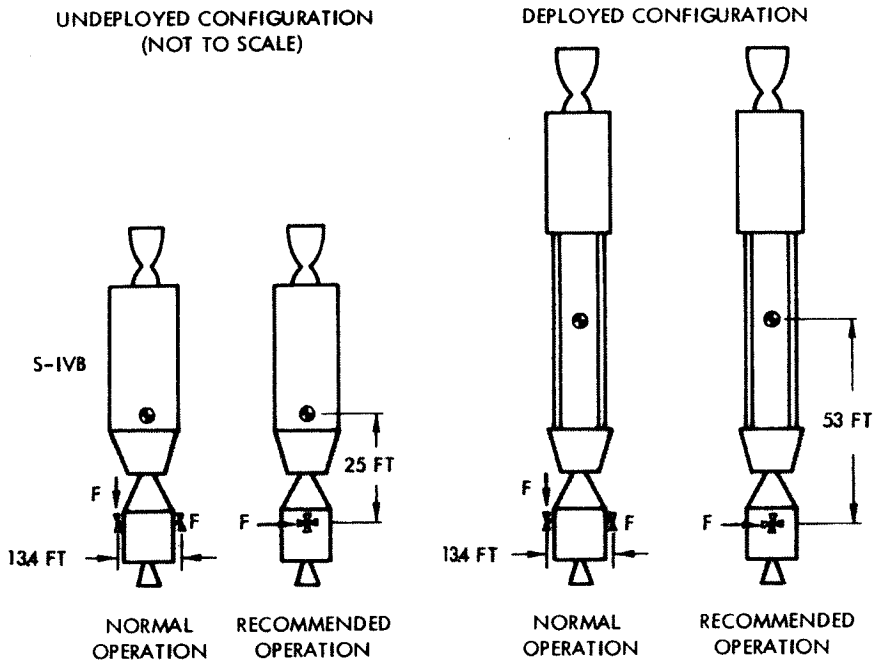

Figure 18. Jet-Arm Comparison for Artificial g Spinup

Table 25. Artificial Gravity Spin Profile

Acceleration (g)	Radius (ft)	Spin Rate (rad/sec)	Angular Momentum (slug-ft^2/sec)
1/6	20	0.519	0.596×10^6
1/6	35	0.392	1.2×10^6
1/6	50	0.328	1.956×10^6
3/5	35	0.745	2.28×10^6
3/5	50	0.622	3.71×10^6
1	50	0.804	4.79×10^6
3/5	50	0.622	3.71×10^6
3/5	35	0.745	2.26×10^6
1/6	50	0.328	1.956×10^6
1/6	35	0.392	1.2×10^6
1/6	20	0.519	0.596×10^6

The rotational mode of operation for this mission will present various problems. For example, it probably will be necessary to mechanize an automatic rotation mode. The available experiment definition data for this mission do not indicate a need for directional control during the rotatior phase. Therefore, this mode would essentially take the form of three-axis rate control. Pitch and roll rates would be nominally controlled to zero and the yaw rate controlled to a specified value. Deadbands would be used to minimize propellant consumption. If it is desired to use the standard flight director attitude indicator for display of rotation rates, scaling changes will be necessary. It is recommended that further effort be devoted to analysis of the rotational phase dynamics and definition of the hardware modifications necessary to provide an automatic rotation control mode.

Flight 518

The solar panels included in Flight 518 as a portion of Experiment1405 result in several alternatives for vehicle design and operation during the sun

orient mode. This mode consists of 789 hours during which the panels are controlled to obtain the maximum solar radiation. Both fixed and single degree-of-freedom panels were examined. For the movable panel case, the the two operational alternatives shown in Figure 19 were considered. In one case, the vehicle longitudinal axis is held along the local vertical and panel control is maintained by rolling the vehicle and rotating the panels. In the second case, the vehicle longitudinal axis is maintained in a plane perpendicular to the local vertical and panel control results from yawing the vehicle about the local vertical and rotating the panels.

The three types of fixed panels evaluated are shown in Figure 20. In Types 1 and 2, no attempt was made to minimize aerodynamic torque effects. For fixed panels, it is necessary to maintain the vehicle in an inertial attitude hold mode for correct panel orientation. During inertial hold, both gravity gradient and aerodynamic disturbance torques act on the vehicle to varying degrees. For Type 3 panels, aerodynamic torques are minimized by attaching the panels to the vehicle so that the center of pressure of the panels coincides with the center of pressure of the vehicle cross-section area. This design feature minimizes aerodynamic disturbance torque effects. Table 26 is a summary of the reaction propellant requirements for the three fixed-panel configurations for the 789-hour sun orient period.

Table 26. Fixed Panel Reaction Propellant Requirements

Vehicle	Propellant Required (pounds)		
	Type 1	Type 2	Type 3
518-C	1040	1040	570
518-D	1230	1230	680

Movable panels are attached to the rack with the axis of rotation perpendicular to the longitudinal vehicle axis. For the local vertical orientation, gravity gradient torques are minimized. Aerodynamic disturbance torques can become excessive, however, because of the distance between panel center of pressure and vehicle center of gravity. In the local horizontal mode, gravity gradient torques again are minimal and aerodynamic torque effects are less than in the local vertical mode. Table 27 is a summary of the reaction propellant requirements for the movable panel operational modes.

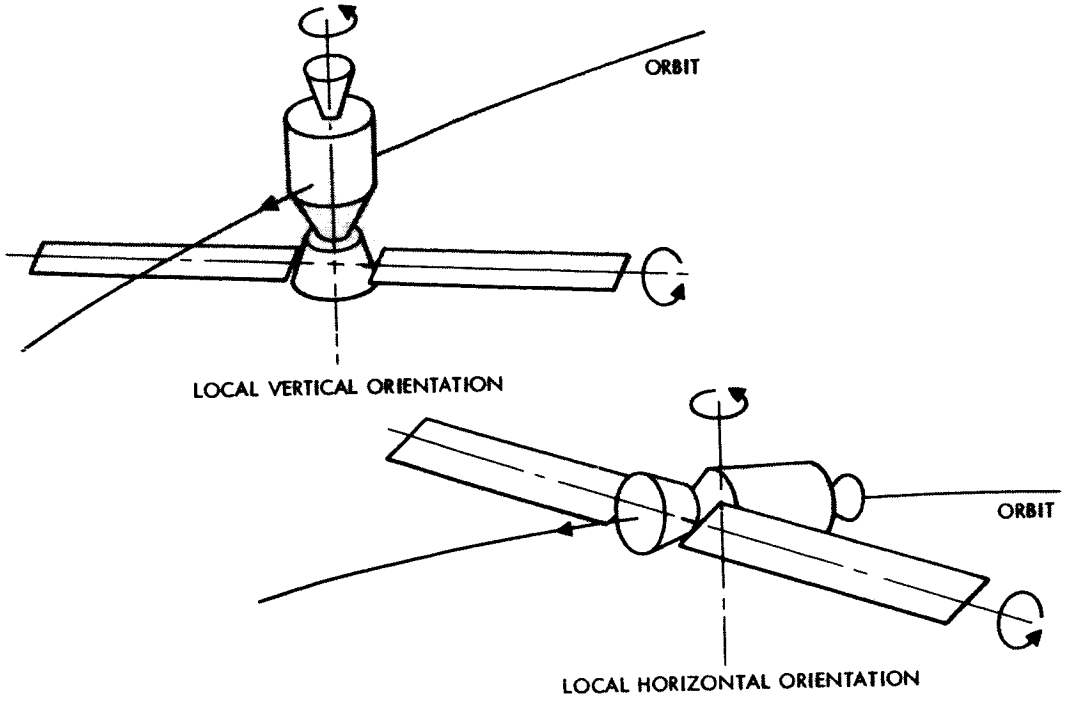

Figure 19. Movable Panel Operational Modes

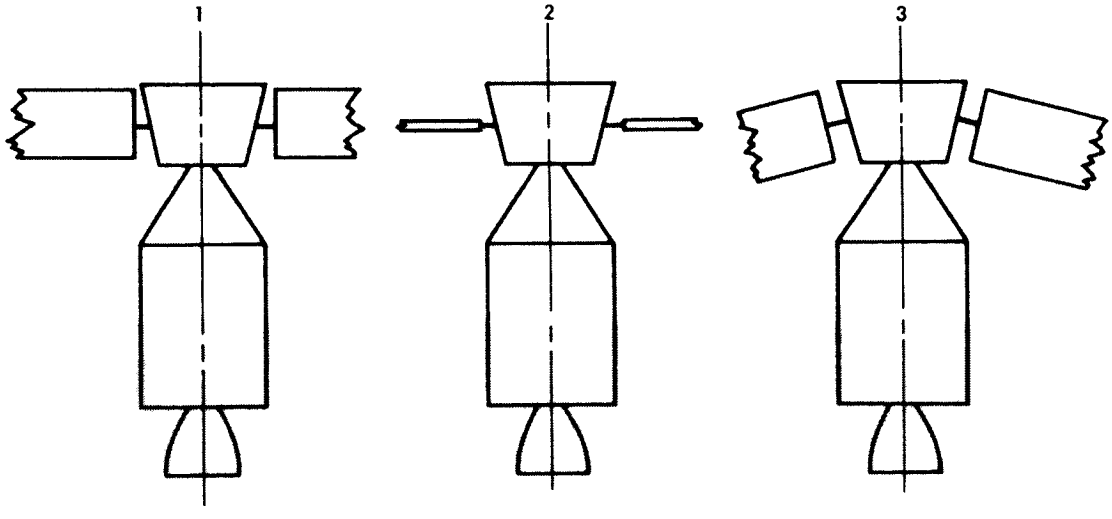

Figure 20. Fixed Panel Design Alternatives

Table 27. Movable Panel Reaction Propellant Requirements

Vehicle	Propellant Requirements (pounds)	
	Local Vertical	Local Horizontal
518-C	1148	487
518-D	1876	511

From a comparison of Tables 26 and 27, movable panels and the local horizontal operational mode are recommended for Flight 518 configurations.

The preliminary reaction propellant evaluation for Flight 518 resulted in propellant requirements considerably in excess of the amount that could be provided using the LEM reaction propellant tanks. Elimination of the 19 spin-profiles associated with Experiments 0101 and 0102 resulted in propellant savings of 2070 pounds and 2570 pounds for the C and D configurations, respectively. Further restriction on the number of maneuvers associated with the experimental program resulted in the reduction of the reaction propellant requirement to the allowable amount (i. e., 2280 pounds).

Although a total propellant requirement of less than 2280 pounds can be attained with a restriction on the number of maneuvers performed during orbit, it is necessary to determine the types of maneuvers so that the propellant can be correctly distributed among the four quads. During the experimental program, the predominant vehicle orientation is the local horizontal mode with the longitudinal spacecraft axis parallel to the earth's surface and in the orbital plane. To maneuver to targets of opportunity from this orientation, it is necessary to yaw and roll the spacecraft only. If propellant estimates are made utilizing only yaw-roll maneuvers, a limitation on the number of such maneuvers is encountered. This occurs because only 1140 pounds of propellant can be utilized for yaw control. When all maneuvers are performed by yaw-roll, the total yaw propellant including maneuvers, ascent and descent, orbit keeping, and ullage, exceeds 1140 pounds. Instead of performing yaw-roll maneuvers, however, it is possible to achieve the same orientation effect with a roll-pitch-roll maneuver. Using the channel disable capability during roll maneuvering, this maneuver can be performed with propellant from the pitch quads only.* The following discussion is an explanation of the method used to determine the respective number of yaw-roll and roll-pitch-roll maneuvers that can be performed during the experiment program.

*It is assumed that roll maneuvers utilize the pitch quads only.

First, the propellant required in the pitch and yaw quads during all mission phases and for all operations except the experiment-related orientation maneuvers was determined. This value was subtracted from 1140 pounds. The remaining propellant is used to determine the number of manual yaw maneuvers associated with each yaw-roll maneuver. This is also the number of yaw-roll maneuvers that can be performed. This value is multiplied by the propellant quantity per roll maneuver and the result added to the mission pitch quad requirements.[1] The resulting pitch quad value is subtracted from 1140 pounds and the remainder divided by the propellant quantity per manual roll-pitch-roll maneuver to give the number of such maneuvers that can be performed. The procedure for Flight 518C is as follows:

	Pitch (and Roll)	Yaw
Ascent and descent	85	71
Ullage	39	39
Orbit keeping	68	68
Panel control	0	330
Attitude hold	320	287
	512 pounds	795 pounds

Excess available yaw propellant = 1140 - 795 = 345 pounds

Number of manual yaw maneuvers = $\frac{345}{1.05}$ = 329 = yaw-roll maneuvers

Roll propellant from yaw-roll maneuvers = 329 x 0.3 = 99 pounds

Total pitch quad propellant = 512 + 99 = 611 pounds

Excess available propellant = 1140 - 611 = 529 pounds

Number of manual roll-pitch-roll maneuvers = $\frac{529}{2.25}$ = 234

[1] It is assumed that roll maneuvers utilize the pitch quads only.

CONCLUSIONS AND RECOMMENDATIONS

The AES missions impose significantly more severe operational requirements on the G&N system than are included in the Apollo LOR design mission. These increased requirements result in definite reliability penalties. The G&N system reliability can be improved by minor system modifications as incorporated in the Apollo X G&N system to reduce the reliability degradation during non-operating periods. For missions that require greatly increased operating time, however, sparing or other provisions for higher reliability will be necessary. Because of the sensitivity of the G&N system to specific mission requirements, rescheduling of experiments on other flights or reducing experiment frequency will have a marked effect on G&N reliability. The more severe requirements of Flights 518 and AF-1 indicate that it would be beneficial to investigate the use of a new system or operating approach for experiment support on these flights.

From a performance standpoint, the AES requirements are less severe. The major changes would be to add a rendezvous and docking capability to the command module G&N system, and to provide for more precise spacecraft stabilization. The new stabilization requirements (0.1 degree and 0.01 deg/sec) can be met, however, by suitable computer program modification and will not require hardware changes.

During the course of the stabilization and control system study, specific problem areas became evident. It is felt that further effort is necessary to evaluate these problems. The feasibility of using manual control for the rotation phase of Experiments 0101 and 0102 should be investigated. As previously discussed, a man in the loop simulation study should be conducted. An automatic rotational mode should be evaluated for use on Flight 221. Dynamics analyses should be conducted, including spacecraft flexibility effects. An analysis should be conducted to estimate the propellant requirements for spin maintenance and wobble damping. The acceleration and deceleration operations also should be analyzed. The addition of an orbit rate torquing mode should be evaluated as a means of satisfying gross local vertical requirements. The control requirements should be revised to reflect the results of the experiment scheduling work. The areas that can be expected to change include operating times and maneuver requirements. System reliability and propellant requirements also can be expected to vary.

PROPULSION AND REACTION CONTROL SYSTEM

Propulsion system analyses included (1) the determination of the service propulsion system (SPS) and the reaction control system (RCS) requirements and (2) the definition of the modifications required on the Block II systems to satisfy the operational requirements established for Configurations C and D. This section discusses the engineering aspects and design integration of the propulsion and reaction control systems with the various AES concepts performed during the study.

The operational profiles with respect to firing durations, starts, and propellant quantities were determined and are presented graphically during this discussion. These requirements cover the experimental-induced orbital requirements, such as linear accelerations, artificial gravity, reorientations, rendezvous, and docking, and stability and control as well as those required for the basic ascent and descent functions.

The propulsion system study was accomplished in accordance with the guidelines established at the beginning of the study. From the propulsion aspects, these basic guidelines simply implied the following:

1. Configuration 1, D' — Use Apollo Block II propulsion systems without modification.

2. Configuration C — SPS and RCS propellant tanks may be resized as was accomplished in the Apollo X studies.

3. Configuration D — Propulsion system changes are restricted to the RCS propellant tankages.

The basic approach used in this study is to utilize as much of the Apollo-based propulsion system concepts and hardware as possible to minimize modifications. Also, because it has been established by previous studies that the command module RCS will be adequate for the missions under consideration, major emphasis was placed on the evaluation of the SPS and the service module RCS.

SERVICE PROPULSION SYSTEM

This section considers the requirements of the AES missions and presents a brief description of the selected systems.

REQUIREMENTS

Preliminary examination of the propulsion requirements revealed that from the aspects of SPS design, the most constraining factor is orbital storage for mission durations greater than 14 days. On the other hand, the total burn time and restarts required by the AES missions are well within the capability of the Apollo Block II SPS. The SPS propellant requirements for these missions are presented in Figure 21. This figure shows that synchronous orbit Flights 509, 516, and 521 would require the full capacity of the Block II SPS propellant tanks. All other AES flights, including the AF-1 and AF-2 flights, can be handled adequately with tanks sized for a total propellant capacity of 10,000 pounds. The 10,000-pound capacity should provide sufficient margin for propellant tolerance, ullage volume, mixture, ratio shift, and residuals. The use of these smaller tanks would result in a more compact and light weight SPS system and, at the same time, would minimize the problems of propellant-sloshing and zero-g feed associated with extreme off-load conditions.

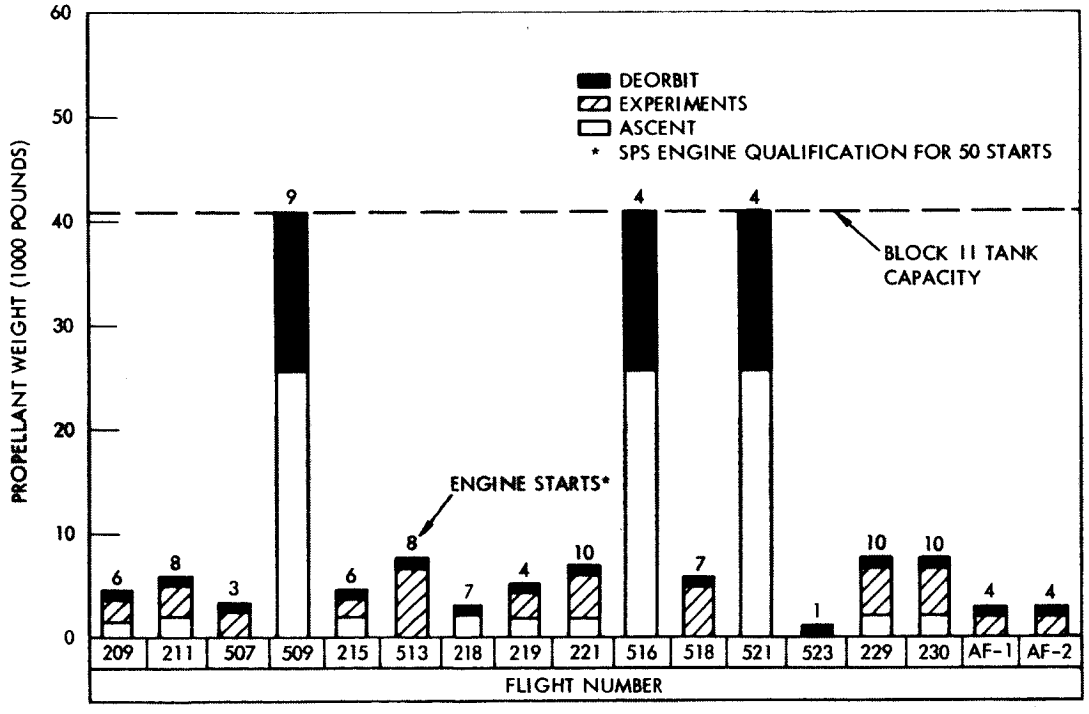

Figure 21. SPS Propellant Requirements

The SPS engine will be utilized to provide the linear vehicle accelerations called for by Experiment 0101. To minimize the propellant requirements and changes in orbital characteristics resulting from this experiment, it is proposed that these engine firings be conducted orthogonally; i.e., perpendicular to the orbit plane. An orthogonal impulse will rotate the orbital velocity vector out of the original plane but will not alter the magnitude of the velocity vector or the orbit alittude of the vehicle. Figure 22 shows the magnitude of changes in orbit inclination with various levels of orthogonal impulses. These data are derived from the equation:

$$\theta = 2 \sin^{-1} \left(\frac{\Delta V}{2 V} \right)$$

where:

θ = plane angle change (degrees).

ΔV = orthogonal impulse (feet per second).

V = orbital velocity (feet per second).

From these data it can be observed that the application of orthogonal impulse has very little effect on the orbit inclination of the vehicle. For example, a 10-second firing of the SPS engine, even at very low gross weight (20,000 pounds), will only tilt the orbit plane less than 1.5 degrees. Also, it should be noted that the linear acceleration experiments should be performed with successive firings that are 180 degrees to one another to keep the vehicle approximately in the same inclination.

SELECTED SYSTEMS

In accordance with the study ground rules, the Apollo Block II SPS is used for all the Configuration 1 flights and the Configuration D' flight without modification. For ease of reference, a brief description of the design characteristics of this propulsion system is presented in the following paragraphs.

The SPS configuration selected for Configurations C and D flights is presented schematically in Figure 23. As shown in the figure, this configuration consists primarily of all Block II plumbing with shortened tanks to provide a total propellant capacity of 10,000 pounds and one helium tank removed. Otherwise, the operations and controls of this system are identical to the Block II design.

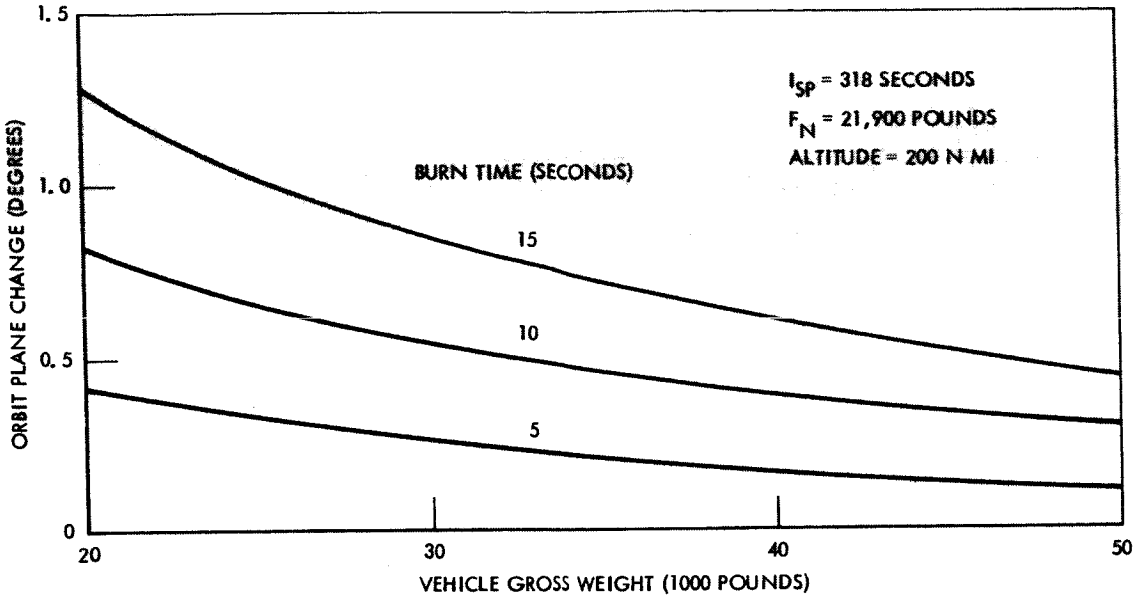

Figure 22. Orbit Plane Change With Orthogonal SPS Engine Firing Versus Vehicle Weight

Service Module Engine

The service module engine is currently being developed by the Aerojet-General Corporation and is identified as model AJ10-137. The bipropellant AJ10-137 engine uses storable propellants, is pressure-fed, provides a fixed thrust of 21, 935 pounds, and is gimbaled for thrust vector control. The engine assembly consists of a thrust chamber, nozzle extension, injector head, propellant valves, gimbal actuators, and a thrust mount.

The injector assembly is fabricated of aluminum and is bolted to the thrust chamber assembly. The face of the injector is semielliptical. The oxidizer and fuel are fed through alternate concentric annuli. The annuli are fed from the periphery through radial manifolds. Several injector patterns are currently being investigated.

The thrust chamber assembly is fabricated of an ablative material that extends out to a nozzle exit area of a 6-to-1 ratio, where the radiation-cooled nozzle extension is attached. The ablative liner material is composed of a Refrasil edge grain material impregnated with a rubber-modified phenolic resin. This material is then wrapped with a thin layer of random-oriented, phenolic-impregnated asbestos that limits the outside wall temperature to 200 F. The outside overwrap consists of epoxy-impregnated glass cloth and finally a filament glass wrap.

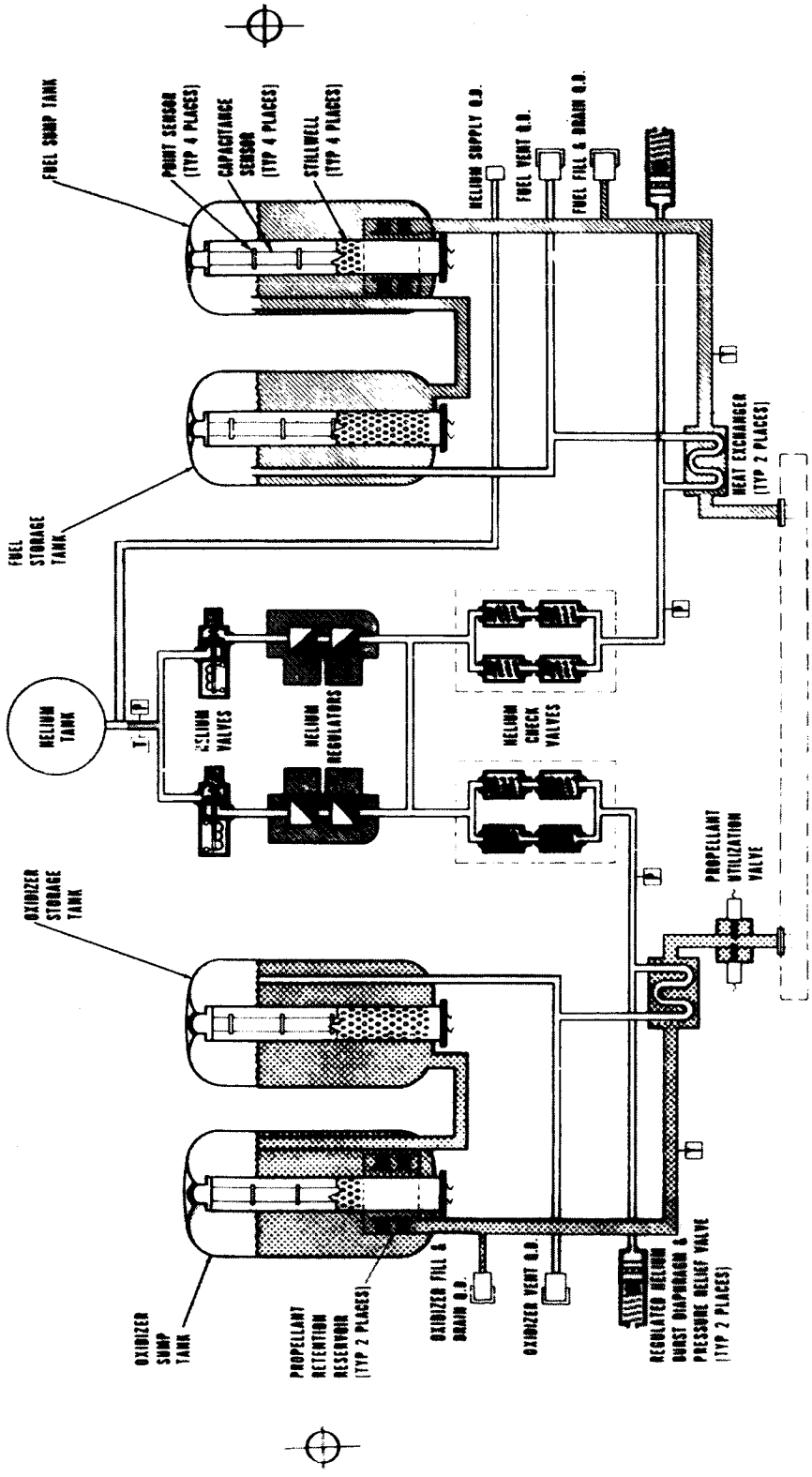
FUEL SUMP TANK
POINT SENSOR (TYP 4 PLACES)
CAPACITANCE SENSOR (TYP 4 PLACES)
STILLWELL (TYP 4 PLACES)
HELIUM SUPPLY G.D.
FUEL VENT G.D.
FUEL FILL & DRAIN G.D.
HEAT EXCHANGER (TYP 2 PLACES)
FUEL STORAGE TANK
HELIUM TANK
HELIUM VALVES
HELIUM REGULATORS
HELIUM CHECK VALVES
PROPELLANT UTILIZATION VALVE
OXIDIZER STORAGE TANK
OXIDIZER SUMP TANK
PROPELLANT RETENTION RESERVOIR (TYP 2 PLACES)
OXIDIZER FILL & DRAIN G.D.
OXIDIZER VENT G.D.
REGULATED HELIUM BURST DIAPHRAGM & PRESSURE RELIEF VALVE (TYP 2 PLACES)

Figure 23. SPS Propellant System (Configurations C and D)

The radiation-cooled nozzle extension is made of columbium and titanium (approximately 0.025 inches thick) with external stiffners for additional strength. The columbium section extends from an area ratio of 6 to 40, where a transition is made to titanium, which extends out to the exit at an area ratio of 60 to 1.

Quad-type propellant valves (series-parallel) are located on the forward engine dome. The valves are pneumatically actuated. Redundant Teflon and butyl rubber seals are used in the valves for internal sealing and for sealing at the inlet and outlet flanges.

A gimbal ring around the throat section of the engine assembly is provided as a thrust mount. Two electromechanical actuators are provided for gimbaling the engine in pitch and yaw (nominal ±7 degrees gimbal angle).

The performance and design characteristics of the AJ10-137 service module engine are summarized in Table 28.

Propellant and Pressurization System

The propellants used in the service propulsion system are nitrogen tetroxide (N_2O_4) as oxidizer and a 50-50 mixture of hydrazine (N_2H_4) and unsymmetrical dimethylhydrazine (UDMH) as fuel. These propellants are earth-storable and hypergolic. The pressurant used in the service propulsion system is gaseous helium. Figure 24 is a schematic of the propellant and pressurization system. The schematic shows that component redundancy has been used frequently to provide the highest practical system reliability. In addition to the component redundancy shown in the schematic, all component and tank seals are double seals or the equivalent.

The propellant system consists of two oxidizer tanks, two fuel tanks, propellant distribution plumbing, fill and pressurant heat exchangers, and a propellant utilization subsystem.

The oxidizer and fuel storage and transfer system are functionally similar; therefore, a description of the oxidizer subsystem is given here and any differences from the fuel subsystem are noted. The oxidizer is contained within two hemispherically domed, cylindrical tanks connected in series by a transfer line and standpipe. One tank is used as a sump tank, and the other tank is used as a storage tank. The Block II propellant tanks are sized to deliver a minimum total requirement of 39,000 pounds of usable propellant. In addition to this minimum requirement, tank space is provided to account for the several inefficiencies, including trapped propellant, oxidizer/fuel mixture ratio shift, propellant loading tolerance, ullage space, and tank

Table 28. Aerojet-General AJ10-137 Apollo Service Module Engine Performance and Design Characteristics

Rate nominal thrust (vacuum), pounds	21,935
Nominal chamber pressure, psia	100
Rated duration (service life), seconds	730
Nominal specific impulse (vacuum) seconds	315
Minimum specific impulse (vacuum), seconds	310
Nozzle expansion ratio	60:1
Fuel	0.5 UDMH +0.5 N_2H_4
Oxidizer	N_2O_4
Ignition	Hypergolic
Nominal mixture ratio	2:1
Engine weight (dry), pounds (wet), pounds	679
Type cooling	Ablative chamber/radiation nozzle/ extension skirt
Transition (ablat/rad)	6:1
Extension skirt material	Titanium-columbium
Type feed	Pressure
Nominal inlet pressure (valve inlet), psia	160
Nominal propellant inlet temperature F	70
Oxidizer inlet temperature range, F Fuel inlet temperature range, F Note: Fuel temperature within ±10 degrees of oxidizer temperature	30 to 135 40 to 145
Gimbal angle range, (nom), cc degrees	±7
Deflection rate, rad/sec Deflection acceleration rate, rad/sec/sec Max. gimbal power requirement (continuous operation), watts	0.30 to 0.35 2.3 to 2.8 2000
Nozzle outside exit diameter inches	100
Overall length (maximum) inches	160
Restarts (minimum)	50
Throttling range	None

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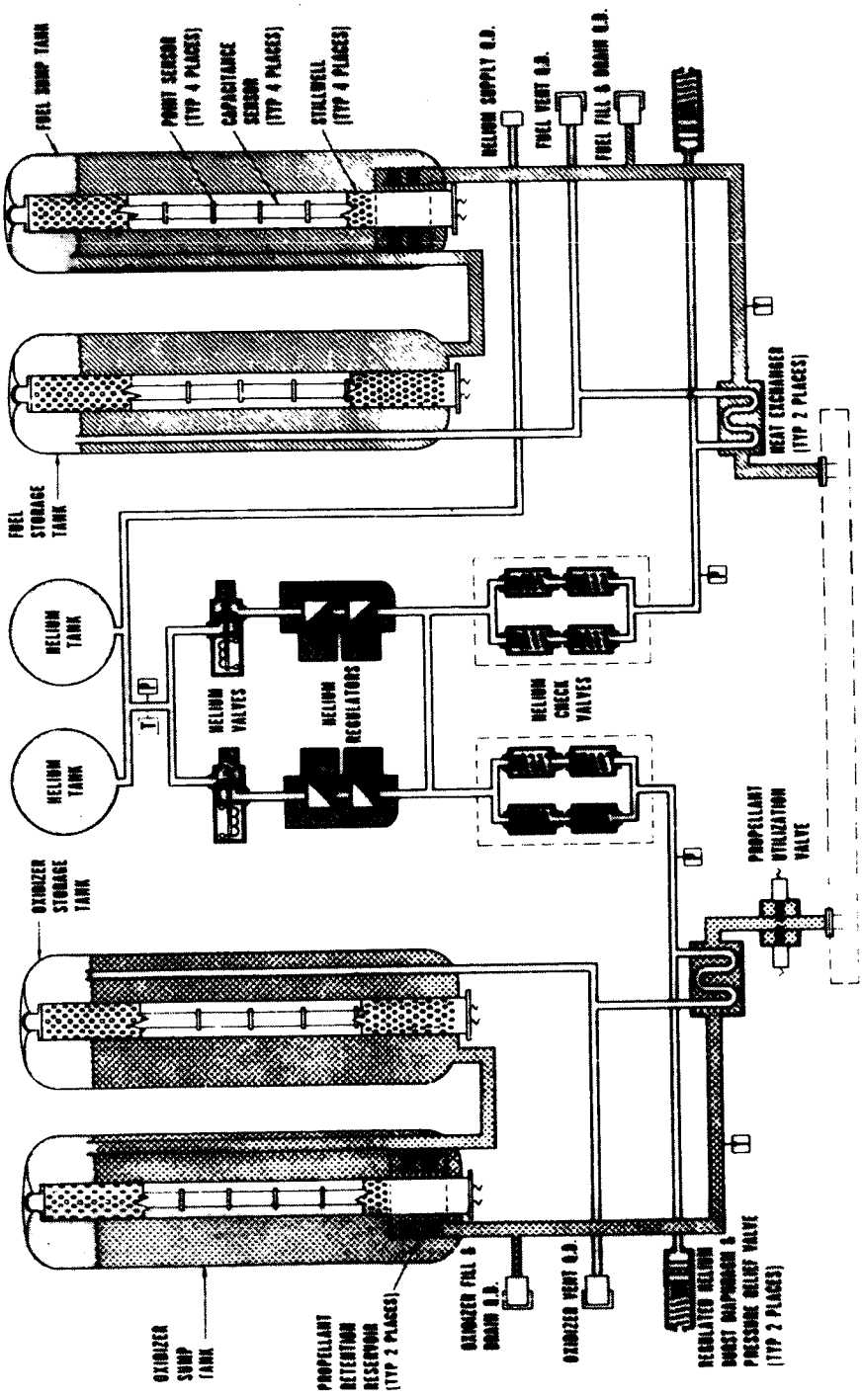

Figure 24. SPS Propellant System (Configuration 1)

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manufacturing tolerance. The tanks do not include positive expulsion devices. To provide for engine starts after periods of zero-g, a can-like assembly is located over the sump tank outlet to retain oxidizer at the tank outlet by surface tension forces and to maintain the feed line full of liquid. In addition, the RCS system provides ullage accelerations prior to each main engine start to settle the oxidizer to the tank bottom. During engine operation, the storage tank is pressurized with helium at the same time that the engine flow control valves are opened. Oxidizer is transferred from the storage tank to the sump tank and then to the engine. The sump tank is maintained full until the storage tank is emptied.

A quantity gauging and mixture ratio control subsystem maintains the desired oxidizer-fuel weight ratio in the propellant tanks during feedout to ensure simultaneous depletion of both propellants. The system also provides propellant quantity information to the crew in the form of prepellant quantity remaining in each tank, total propellant quantity remaining, and the ratio of oxidizer-to-fuel quantity remaining. The propellant quantity remaining is measured by a capacitance probe in each tank. Also, point sensors are located along the probe to provide a backup quantity measurement device. The electronics required and the crew displays for the two measurement devices are completely separated for high reliability. If the oxidizer-to-fuel ratio of the remaining propellant varies from the desired 2:0, three-position propellant utilization valves located in the oxidizer feed line are adjusted by the crew to increase or decrease the propellant oxidizer-to-fuel ratio flowing to the engine as required to return the remaining propellant to ratio of 2:0. The propellant utilization valves are then returned to the nominal position.

A counterflow heat exchanger is installed in both the fuel and oxidizer feed line to heat the helium pressurizing gas to the approximate propellant temperature prior to pressurizing each storage tank. The helium supply is cooled during propellant usage because of the gas expansion within the storage spheres. The use of a heat exchanger reduces the amount of helium required. Also, the heat exchanger prevents the possibility of cold helium entering the storage tanks and causing undesirable pressure increases when the helium is heated by the warmer propellant.

The pressurization system consists of two helium storage spheres, solenoid isolation valves, dual stage pressure regulator assemblies, check valve assemblies, pressure relief valves, a fill valve, and a vent. Helium is stored at 4000 psia in two spheres that are manifolded together.

Helium is isolated within the storage tanks by two solenoid operated valves that operate automatically during the engine start and shut-down sequence. These isolation valves are open only while the main engine is operating. Each valve has a manually actuated override in the crew station for emergency. The valves are installed in parallel for system reliability.

Pressure regulation is accomplished by two pressure regulator assemblies installed in parallel. Each regulator assembly incorporates a primary and secondary regulator in series. The primary regulator reduces the helium pressure to 185 ± 4 psia. The secondary regulator remains open as long as the primary regulator functions properly. If the primary regulator fails to open, the secondary regulator is capable of maintaining a pressure between 181 and 194 psia.

A redundant series-parallel check valve assembly is located in the helium distribution line to each propellant tank to prevent back flow and mixing of the oxidizer and fuel vapors. The check valve assembly consists of four independent check valves installed in a series-parallel combination.

A relief valve assembly, consisting of a relief valve, burst diaphragm, and filter, is provided downstream of each check valve assembly to prevent overpressurization of the propellant tanks. The burst diaphragm is designed to rupture at 220 ± 7 psig. Prior to rupture of the burst diaphragm by excessive tank pressure, the diaphragm prevents propellants from entering the relief valve. The relief valve is designed to open at 232 ± 8 psig and have a minimum reseat pressure of 212 psig.

REACTION CONTROL SYSTEM

The primary effort of the RCS study was to determine the operational requirements and modifications necessary to increase the capability of the Block II system to meet the experimental mission requirements of all the Configuration C and D flights. The operational requirements established were based on the following assumptions:

1. The specific impulse for attitude-hold maneuvers is 150 seconds and 270 seconds for all other maneuvers.

2. All angular maneuvers were made at rates of either 0.2 degrees per second or at 0.05 degrees per second for the AES flights and at rate of 0.2 degrees per second or 0.01 degrees per second for the AF flights.

3. Propellant requirements are neglected for arresting transients caused by such disturbance torques as SPS engine shutdowns, crew motion, solar pressures, etc.

REQUIREMENTS

Figure 25 depicts the RCS propellant required for the various flights. Inspection of this figure revealed that (1) the propellant capacity of the Block II service module RCS will be adequate for all the Configuration 1 Flights 209, 211, 507, 509, 512, and 513 and that (2) the propellant capacity of the Block II system will not be sufficient for the Configurations C and D Flights 219, 221, 518, 229, 230, and AF-1. The total propellant required for the latter flights is approximately three times the total propellant available from the Block II system. This increase in propellant requirement is due to the need for reorientation, artificial gravity, and attitude-hold maneuvers imposed by the experimental missions associated with these flights. Because the basic aim of this study was to minimize modifications, it was decided to increase the propellant supply for the more energetic missions through the use of multiple LEM RCS tanks as was done in the Apollo X study. In this design, two sets of LEM RCS propellant and helium pressurant tanks would be required in each quadrant, as illustrated on Figure 26. This configuration will yield a total usable propellant of approximately 2300 pounds. In this four-independent quadrant system, the pitch and yaw propellant is divided between their two quadrants, and the roll and X-axis translation propellants are divided among the four quads. It

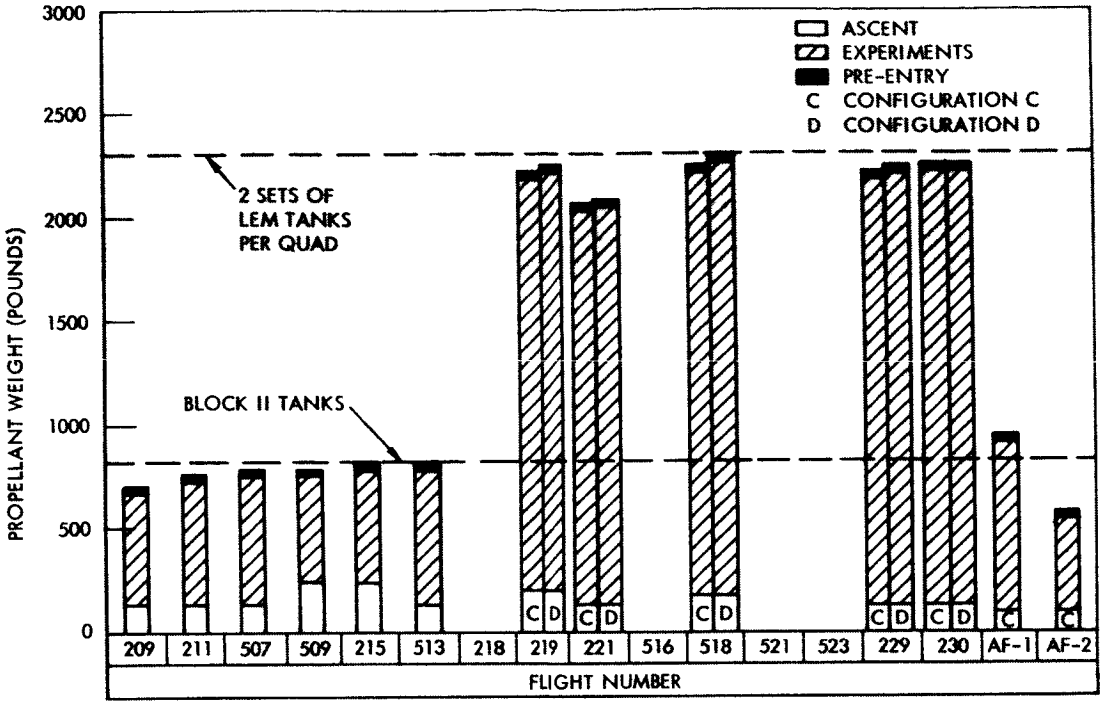

Figure 25. RCS Propellant Requirements

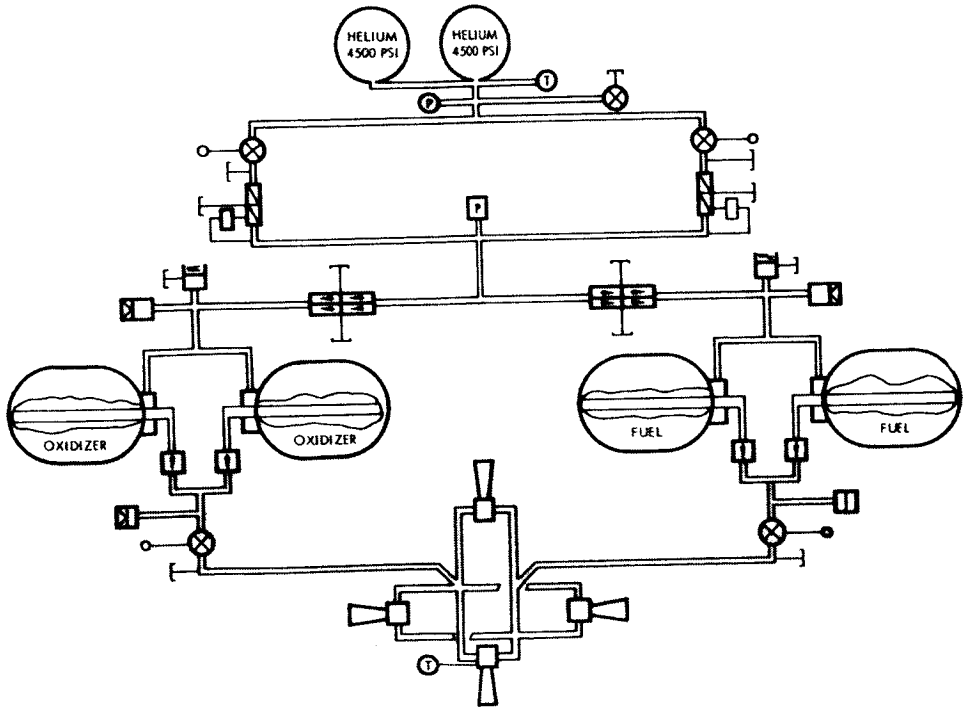

Figure 26. RCS Schematic for Configurations C and D

should be realized that with this type of arrangement there is a potential problem of prematurely depleting the propellant from one quad. Of course, this problem may be circumvented by properly scheduled vehicle maneuvers.

The most stringent requirements imposed upon the RCS are the operation cycles of the yaw jets on Flight 518, as indicated on Figure 27. The estimated starts required for the yaw jets are 21,000 cycles, which total more than twice the specification requirements for the Block II RCS jets. Preliminary study indicates the need for developing an engine with higher restart capability to satisfy the rigid attitude-hold and vehicle reorientation required for the efficient use of solar cells. An alternate solution to this problem would be the use of a redundant set of yaw thrusters that would require considerable redesign of the quad and associated plumbing. The RCS jets on time are shown in Figure 28.

A brief study was conducted to determine the RCS propellant required to bottom the SPS propellant for zero-g engine starts. The time and RCS propellant required to bottom the SPS propellant under zero-g conditions are primarily a function of the ullage force and propellant tank geometry. The time required to bottom the propellant was estimated by

$$t = \left(\frac{2S}{\alpha g_o}\right)^{1/2}$$

and the propellant required was determined by

$$W_p = \frac{Ft}{I_{sp}}$$

where:

t = time to settle propellant

S = distance propellant is moved

α = gravity ratio, acceleration per g

g_o = earth gravitational constant

W_p = weight of propellant

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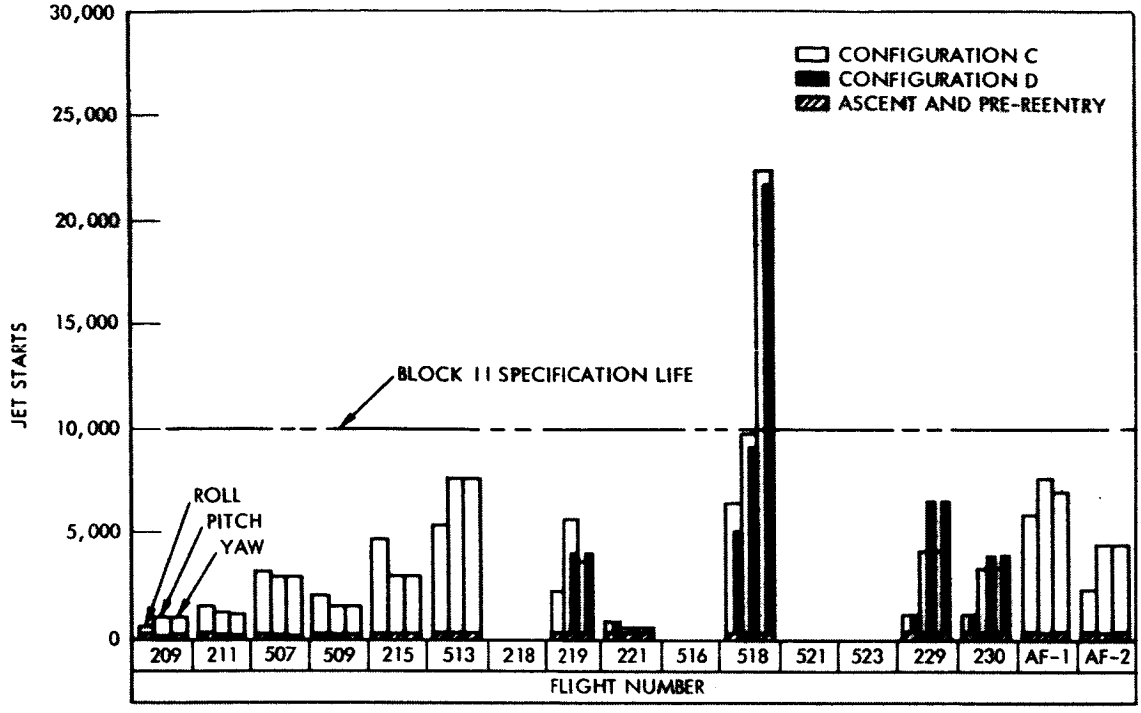

Figure 27. RCS Jet Starts

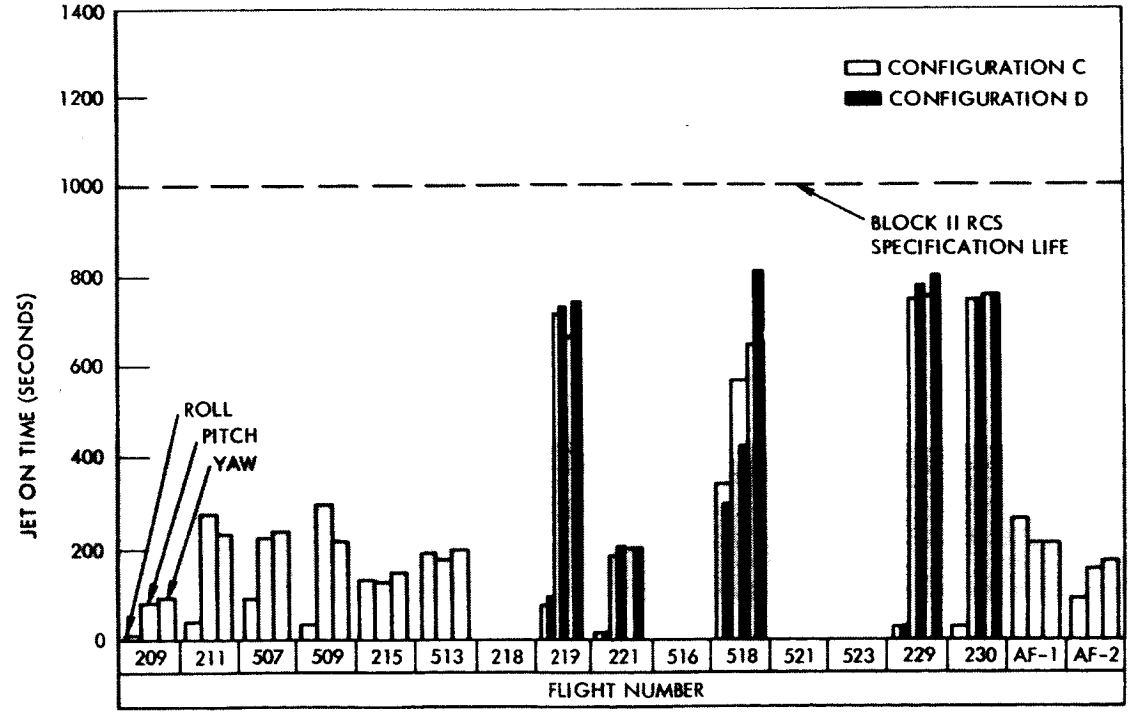

Figure 28. RCS Jet On-Time

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F = thrust

I_{sp} = specific impulse

Figure 29 presents the results of this analysis. The data presented on this figure are computed for the Block II tank configuration with ullage acceleration provided by four RCS jets (a total of 400 pounds of thrust).

SELECTED SYSTEMS

As was done with the SPS, the Block II RCS will be adopted for all the Configuration 1 flights without redesign. The following paragraphs discuss the design and operational characteristics of these systems.

The Apollo spacecraft is provided with two completely independent reaction control subsystems; the command module and the service module reaction control systems. These subsystems are similar in that both are pressure fed, utilize hypergolic propellants, and contain total redundancy of components. The primary purpose of these subsystems is to provide attitude control and stabilization.

Specifically, the command module reaction control system accomplishes roll maneuvers and provides three-axis control of the module after separation from the service module for entry orientation. The service module reaction control system can also provide flight stabilization, attitude control, minor vehicle velocity adjustments, artificial gravity, docking, and departure. Both subsystems operate in response to control signals from the stabilization and control system or to a secondary electrical circuit connected to manual override controls.

Configurations 1 and D'

The basic reaction control system consists of a pressurization system, a propellant supply and distribution system, and the rocket engines.

Each pressurization system includes a helium supply bottle, parallel redundant isolation valves, parallel series pressure regulators, parallel series redundant check valves, a provision for propellant supply pressurization and pressure relief, and servicing and checkout equipment. Because the command module reaction control system is used only during reentry, an additional squib-actuated isolation valve is used to seal off that system prior to reentry to minimize or eliminate propellant and pressurant leakage. Helium is used as the pressurant in both systems.

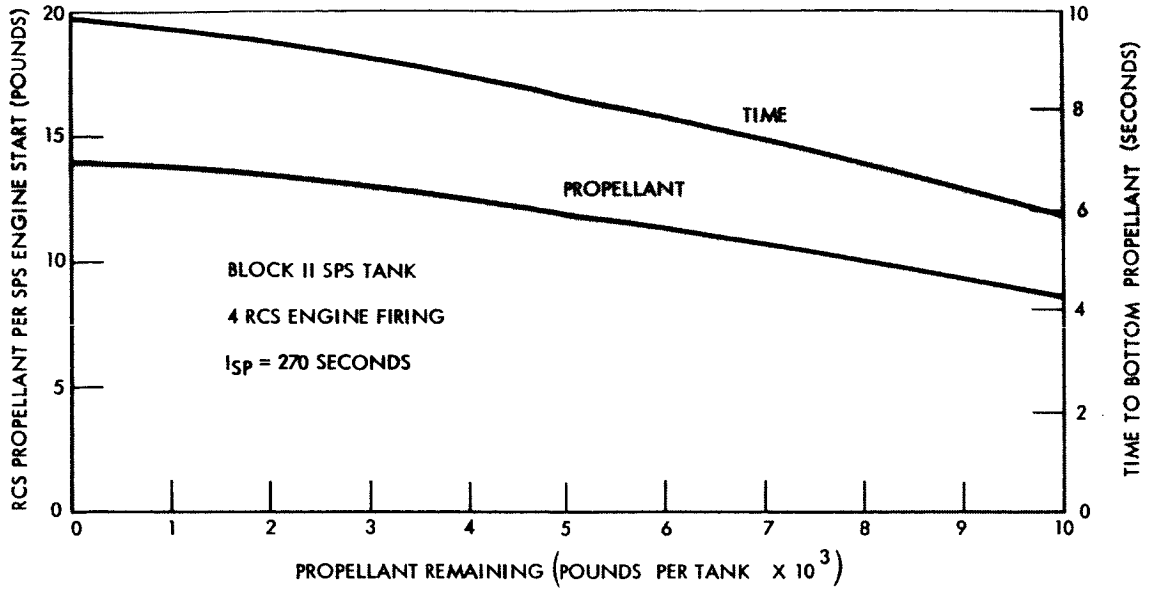

Figure 29. SPS Propellant-Settling Requirements

Each service module RCS package contains one spherical pressure vessel that is cradle-mounted to eliminate the possibility of detrimental skin flexures. Each helium tank contains 0.205 cubic feet of helium gas, initially stored at a nominal 4000 psia and 70 degrees F. Limit pressure is 4500 psia; proof pressure, 6000 psia; and burst pressure, 6750 psia. Maximum allowable leakage at the proof pressure is 4×10^{-6} pounds per hour. Two helium isolation (shut-off) valves are placed in parallel just downstream of each helium tank. These solenoid-operated valves are crew-controlled. The high-pressure helium, after passing through the isolation valves, enters the pressure regulators that reduce the pressure to approximately 180 psig. The general arrangement of this system is shown on Figure 30.

The regulation assemblies are placed in parallel, each downstream of and in series with the isolation valve. One regulator assembly incorporates a primary and secondary regulator in series. The primary (upstream) regulator in the series reduces the high upstream pressure to the normal downstream operating pressure.

The secondary regulators do not operate until some malfunction of the primary regulators cause a rise in the secondary regulators inlet pressure, in which case the secondary regulators operate to maintain the required regulator outlet pressure.

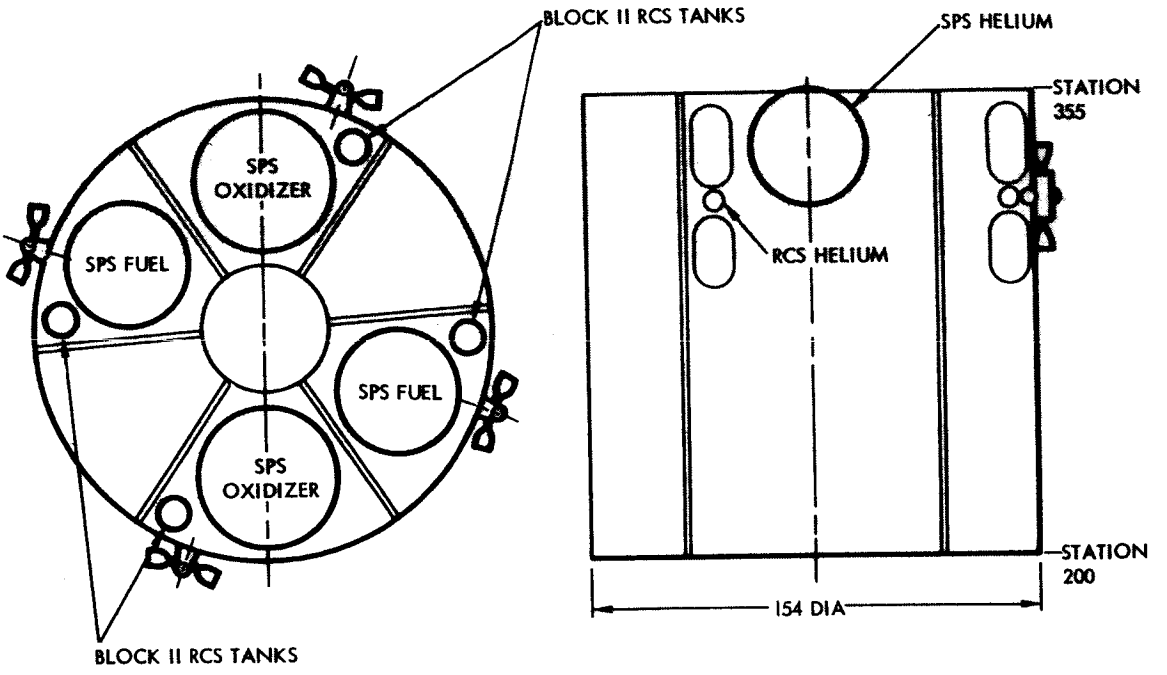

Figure 30. Service Module RCS for Configuration 1

Flow is divided downstream of the regulators, and separate lines supply helium to the fuel and oxidizer tanks. A parallel series redundant check valve assembly is positioned in each of these individual helium distribution lines to prevent cross-flow and mixing of fuel and oxidizer vapors. In the command module system, helium interconnect lines located downstream of the check valves are used to allow for a cross-over system possibility between systems A and B. Pressurant from either system can be used in its cosystem through the operation of a squib-actuated valve.

Each propellant supply distribution system includes positive expulsion fuel and oxidizer tanks, fill and vent connection, propellant shut-off valves, and provisions for system checkout.

The systems use cylindrical tank assemblies with hemispherically domed ends, triple-ply bladder positive expulsion devices, and support assemblies for both the fuel and oxidizer tanks. The fill valve is located just downstream of the tanks, and the vent valve is in the helium line just upstream of the tank.

The volume of each tank, with the expulsion device installed for each system, provides the following capacities for oxidizer and fuel at a temperature of 65 F:

Command module (two independent systems)

Oxidizer - N_2O_4	89.2 pounds
Fuel - MMH	45.2 pounds

Service module (four independent systems)

Oxidizer - N_2O_4	138.1 pounds
Fuel - 50-50 blend UDMH—N_2H_4	69.7 pounds

With the tanks oriented in any position and the propellants at any temperature from 40 F to 120 F, the expulsion device expels 99 percent of propellant capacity.

Helium leakage through the expulsion device into the propellant compartment does not exceed 2×10^{-6} pounds per hour at any pressure up to the proof pressure.

With the exception of the propellant system shut-off valves, all valving of the propellants is accomplished by the reaction control engines. In the command module, an intersystem connection, including a squib-actuated valve, is made to allow for a cross-over between systems A and B, much like that of the helium system.

The general method of operation of the reaction control system would start upon opening of the propellant isolation valves. The propellant flows directly from the pressurized propellant storage tanks to the engines. The actual firing of a particular reaction control system engine is controlled by the engine propellant valve.

Configurations C and D

For the flights assigned to Configurations C and D, the Block II service module RCS was found to be inadequate from the standpoint of propellant capacity. Therefore, an alternate RCS configuration is proposed for these latter flights. As shown on Figure 31, this alternate system utilizes two sets of LEM RCS and helium tanks in each quadrant. This design will yield a total usable propellant quantity of approximately 2300 pounds.

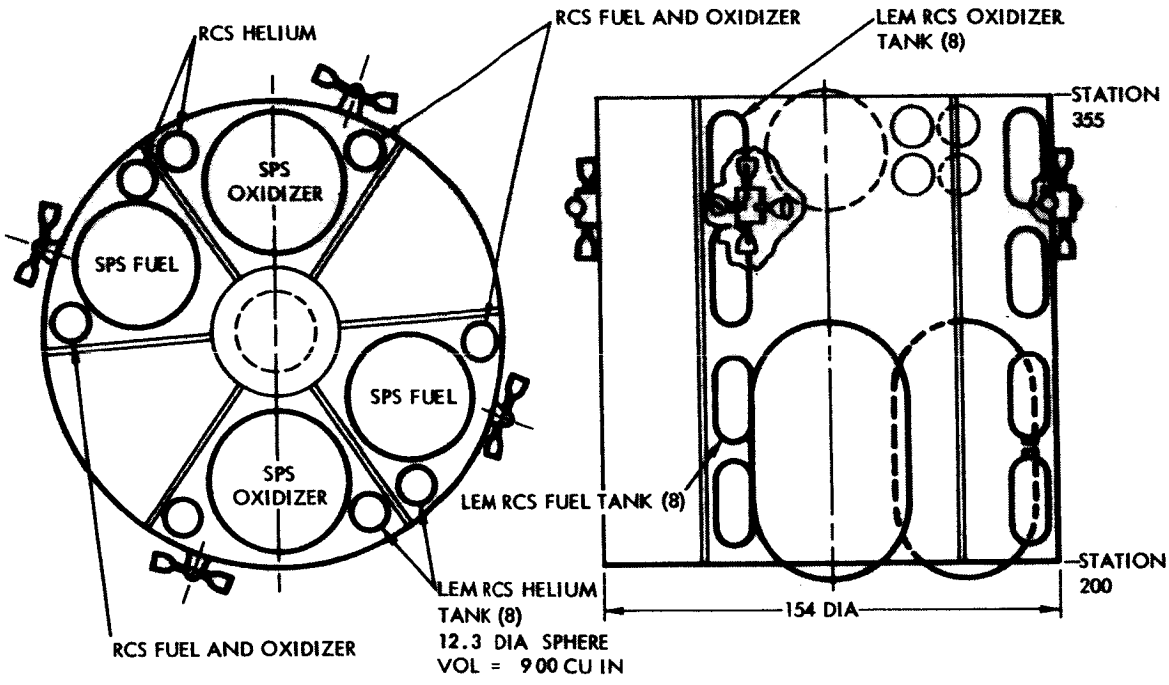

Figure 31. Service Module RCS for Configurations C and D

Table 29 presents a summary of the service module RCS design characteristics.

Table 29. Service Module RCS Design Summary

Parameter	Configuration		
	1	C	D
Chamber type	Radiation-cooled	Radiation-cooled	Radiation-cooled
Thrust (pounds)	100	100	100
Chamber pressure (psia)	94	94	94
Nozzle area ratio	40:1	40:1	40:1
Specific impulse (seconds)	300	300	300
Propellants	N_2O_4/50-50	N_2O_4/50-50	N_2O_4/50-50
Mixture ratio	2:1	2:1	2:1
Duration (seconds,)	1000	1000	1000
Total propellant weight, (pounds per quad)	207.8	613.6	613.6
Fuel weight (pounds)	69.7	204.6	204.6
Oxidizer weight (pounds)	138.1	409.0	409.0
Fuel tank volume (cubic feet)	1.232	3.60	3.60
Oxidizer tank volume (cubic feet)	1.528	4.55	4.55
Pressurant tank volume (cubic feet)	0.205	1.04	1.04

CONCLUSIONS

Based on the results of this study, the following observations and conclusions were made:

SERVICE PROPULSION SYSTEM

The Apollo Block II system will be adequate for all Configuration 1 flights. It should be noted, however, that Flights 209, 211, 507, 215, and 513 may be accomplished more efficiently with a reduced propellant tank size.

For the flights assigned to Configuration C, only Flights 516 and 521 would require the full capacity of the Block II tanks. All other flights may be accomplished with smaller size propellant tanks.

REACTION CONTROL SYSTEM

The Block II service module RCS will be sufficient for all flights assigned to Configuration 1 without modification.

Two sets of LEM RCS tanks per quad will be required to accommodate the increased propellant quantity required for the Configuration C flights.

Flight 518 imposed very stringent operational requirements on the yaw thrusters in terms of starts. It is anticipated that considerable design improvement and development testing would be required to make the Block II RCS engine suitable for the 21,000 starts required by this flight.

Limited study indicates that the thermal control system currently being considered for Apollo will not be adequate for the AES missions. Increased orbital duration and rigid orientation and control for these experimental missions will necessitate certain modifications to the basic Apollo system, the extent of which cannot be determined without more detailed thermal analysis.

POWER SYSTEM

The achievement of the AES program objectives is greatly dependent on the availability of power, both for the loads caused by experiment operation and to acheive the 14-day, 30-day, and 45-day mission duration goals. The momentary accomplishment of any one particular function or experiment requires that power be available at the right time and in the right form and quality, while repeated or sustained operation for long mission durations is dependent on energy (rather than power) availability. Both of these aspects were analyzed during the study in conjunction with vehicle definition and experiments integration activities to arrive at the most feasible method of meeting program objectives within the constraints of the study ground rules. These ground rules dictated that Configurations 1 and D use the Apollo Block II CSM with as little modification as possible. Additional power provisions as required were to be located in the external device. One 30-day Flight (211) was scheduled relatively early in the program and, therefore, did not allow as much time for subsystem improvements as the other Configuration D flights. This special case (Configuration D') was treated as a separate configuration in the study.

Configuration C was based on the Apollo Systems Utilization Studies (Apollo X) and did allow incorporation of modified and improved subsystems in the CSM.

REQUIREMENTS

The AES program is based on use of Apollo technology, equipment, and even the hardware to accomplish functions other than the original goals of the Apollo program. Thus, the design flexibility is limited, and the study requires careful consideration of the ease of modification of the Apollo vehicle and subsystems in addition to system optimization to satisfy the requirement. The power loads and load profile for each flight will be unique to that flight; thus, the power system design requires a great flexibility.

The power loads for the various configurations were divided into three categories; (1) those required for boost and return, (2) the housekeeping loads required for normal operation of the spacecraft but without attitude hold, guidance, and processing of data other than that of the spacecraft itself, and (3) the loads caused by performing a particular experimental function.

[redacted]

BOOST AND RETURN LOADS

Boost and return loads are predicted with a relatively high degree of confidence since they represent simultaneous operation of most of the Apollo equipment. Peak loads occur as a result of higher cabin pressure in the atmosphere, use of controls, gimballing the SPS engine, etc., and will differ in length and frequency depending on the trajectory. Table 30 shows the average power loads, including inverter and power distribution system losses, for low altitude and synchronous earth orbits. The synchronous average is based on continuous use of the guidance and navigation system and, thereby, represents a more conservative case than would really be required if energy was marginal, since the system could be turned to Standby during part of this mission phase, if necessary.

HOUSEKEEPING LOADS

The power loads required to keep the spacecraft operating on a normal basis are shown in Table 31. Again, there is a variation between low altitude and synchronous orbits, this time caused by use of the communication system power amplifier in synchronous orbits. The housekeeping loads primarily consist of spacecraft operation functions such as fuel cell parasite power, cryogenic storage system fans and heaters, environmental control system pumps and blowers, and spacecraft monitoring functions, including continuous measurements, some controls and displays, and some data processing. Communications functions include continuous use of the central timer, signal conditioner, recorder, audio center, and data transmission to a ground station on the average of four minutes per orbit, including equipment warmup. Service module RCS power includes propellant heaters on two of the four quadrants to maintain the temperature within limits without rolling or barbecuing the vehicle to distribute heat. The guidance and navigation power is the standby level and does not include any G&N system operation, since such operation would be caused by an experimental requirement rather than being necessary for spacecraft operation.

Analysis of the housekeeping loads show that most of the power is used continuously rather than being cycled. For example, fuel cell parasite loads, ECS, measurement, G&N standby, continuously operating data processing equipment, and low level illumination represent an almost constant base to the power load profile, unless an emergency condition arises where the non-essential loads must be dropped. The most significant cyclic loads are caused by turning on of the cryogenic oxygen tank heaters randomly (pressure controlled) and by the load of data transmission that can be predicted by the orbit epheremis. Service propulsion system engine firing is entirely predictable and astronaut-controlled, and is done only a few times during the mission. The difference between Block II and Apollo X loads is primarily due to operating two fuel cells at one time rather than three, and to changes in data processing and G&N equipment.

Table 30. AES Housekeeping Power Loads (Boost and Return)

System	Block II				Apollo X			
	AC		DC		AC		DC	
	Low Altitude	Synchronous	Low Altitude	Synchronous	Low Altitude	Synchronous	Low Altitude	Synchronous
Communications	180	275	53	71	247	342	56	74
Controls and displays	99	99	99	99	99	99	99	99
Crew systems	2	7	2	7	2	7	2	7
Cryogenic storage	24	24	6	25	24	24	6	76
ECS	386/256.	386/256	59	59	386/256	386/256	59	59
G&N	31	31	505	505	31	31	505	505
Fuel cells	366	366	12	12	244	244	12	12
Illumination			170	170			170	170
Measurement			112	112			112	112
SPS*			1400	17			1400	22
SM/RCS			50	50			50	50
Stabilization control	152	43**	124	35*	152	43*	124	35**
Total	1240/ 1180	1231/ 1101	2592	1162	1185/ 1055	1176/ 1046	2595	1221

Average power required***	Block II	Apollo X
Low altitude, Low inclination (LIEO) and Low altitude, Polar or 81 degree (EPO)	3000	2900
Synchronous orbits	2780	2750

*5600 peak, 1400 minimum, while engine is being checked or fired. Average value depends on mission.

**Based on use 2/7 of the time

***Based on 72 percent inverter efficiency and 94 percent d-c power distribution efficiency.

Table 31. AES Housekeeping Power Loads (In-Orbit)

System	Block II				Apollo X			
	AC		DC		AC		DC	
	Peak	Average	Peak	Average	Peak	Average	Peak	Average
Communications								
SEO	275	114	71	30	342	116	74	30
LIEO AND EPO	180	111	53	30	247	114	56	30*
Controls and displays			99	7			99	7
Crew systems			18	7			18	7
Cryogenic storage	100	24	356	25	100	24	356	76
ECS	256	256	59	59	256	256	59	59
G&N	31	0	505	69	31	0	505	49**
Fuel cells	366	366	12	12	244	244	12	12..
Illumination			170	71			170	71***
Measurement			112	112			112	112
SPS	50		5608	0	50	0	5608	0
SM/RCS			210	50			210	50.
Stabilization control	217	0	178	0	217	0	178	0
Total								
SEO		760		442		640		473
LIEO and EPO		757		442		638		473
Gross power required...								
SEO				1526				1392
LIEO and EPO				1521				1389

*Signal conditioner, recorder, audio center and timer ON continuously. All other systems, 4 minutes per orbit. Apollo X has added C-band transponder. HF transceiver, VHF/AM transceiver, VHF beacon, and high gain antenna not used.

**IMU on standby continously in Apollo X. IMU plus computer on continuous standby in Block II.

***Apollo average use rate assumed

.*Includes heating of two SM/RCS quadrants continously

..*Two fuel cell modules operating in Apollo X, three modules operating in Block II

...*Uses 72 percent inverter efficiency and 94 percent d-c power distribution efficiency

EXPERIMENT LOADS

The power required to operate the experiments and the additional usage of spacecraft equipment caused by the experiments is by far the most uncertain portion of the total power load estimate. Experiment definition and integration was proceeding concurrently with other activities during this program and, as a result, these load estimates fluctuated. These loads also varied from one flight to another since each flight has a different set and usage of experiments. Under these conditions, some engineering assumptions and relatively gross calculations had to be made for spacecraft design to proceed and to determine the basic capabilities of the system.

Each experiment was analyzed to determine its power requirements and its demands on spacecraft subsystems. Each experiment was also assigned to a particular flight but not necessarily scheduled for a particular time. The power required to operate auxiliary equipment to support the experiment (including SCS, data processing, G&N, controls and displays, etc.) was also determined for each experiment, without regard to scheduling. However, upon totaling the mission power requirements for a particular function, it was found that this total sometimes exceeded that obtained by continuous use of that equipment throughout the mission. In such cases, the continuous-use maximum was used in further analysis. It was realized that this procedure normally resulted in over-conservative numbers due to time overlapping of requirements from more than one experiment. (For example, if two experiments require 8 hours of G&N usage, scheduling them together reduces this requirement to one-half.) However, this method of analysis, did provide worst-case conditions (assuming the basic experiment definition is correct) and was useful in determining if the spacecraft could be designed to accommodate the experimental requirements. Results of this analysis for each of the four configurations are shown in Table 32. This table shows that all Configuration 1 flights are energy-limited to less than 14 days unless an additional energy source is provided in the external device. All C and D configurations are adequate with the possible exception of Flight 219 which is marginal , but can be accommodated by slightly changing the cryogenic loading procedure on that flight.

To determine the effect of computer scheduling of experiments on the power requirements analysis, Flight 215 was given priority effort to be analyzed before other flights. The variable loads caused by experiment operations was printed in load profile form by a CRT printout, and the total integrated area under the curve was calculated by the computer. The results indicate that charging G&N, SCS, data processing, etc. to each experiment individually without regard to scheduling results in a significant over-design. By scheduling experiments so that these subsystems provide for several experiments at once, and by rejecting those experiments that cannot be scheduled on that run, the energy requirement is considerably reduced.

Table 32. Power Load Vehicle Capability Analysis Summary

Configuration	Mission	Boost & Return (kwh)	Housekeeping (kwh)	Experiments (kwh)	Freezer (kwh)	Rack Displays (kwh)	C&N (kwh)	SCS (kwh)	Experiment Data Processing (kwh)	Total Requirements (kwh)	Service Module Capability (kwh)	Additional Required (kwh)	Available Energy from Rack (kwh)	Mission Duration (days)	Remarks
1	209	12	512	5	17	34	24	4	80	688	620	68	68 / 0	14 / 12.3	For 14 days, use six 450 A-H batteries in Rack
D'	211	12	1090	36	36	22	56	3	103	1358	620	738	920	28.4	H_2 must be used or vented according to a specific schedule
1	507	18	512	25	17	20	35	7	45	679	620	59	59 / 0	14 / 12.8	Requires five, 450 A-H batteries
	509	50	485	9	17	13	25	5	37	641	620	21	21 / 0	14 / 13.5	Requires two, 450 A-H batteries
	215	2	512	35	17	34	49	9	98	766	620	146	146 / 0	14 / 11.3	Requires twelve, 450 A-H batteries
	513	18	512	14	17	34	48	15	70	728	620	108	108 / 0	14 / 11.9	Requires nine, 450 A-H batteries
D	219	12	1543	354	54	108	101	23	326	2511	620	1891	2018	45	At this use rate, SM tanks last 11.3 days. ∴ Apollo X Tanks can be loaded to 84-pound/tank without venting
	221	18	1543	42	54	53	64	145	134	2053	620	1433	1830	45	
	518	18	1543	208	54	88	112	119	167	2309	620	1689	1830	45	Solar cell experiment produces power
	229	12	1543	360	54	108	64	28	277	2446	620	1826	1830	45	
	230	12	1543	355	54	108	65	16	258	2411	620	1791	1830	45	
	AF-1	12	1543	317	*	*	148	34	*	2054	620	1434	1830	45	
	AF-2	12	1543	203	*	*	77	28	*	1863	620	1240	1830	45	
C	219	12	1500	359	54	108	101	23	326	2478	2440	38		44.4	Can probably be done by high-pressure loading the H_2 tank
	221	18	1500	42	54	53	64	145	134	2010	2440			45	
	518	18	1500	208	54	88	112	119	167	2266	2440			45	
	229	12	1500	360	54	108	64	28	277	2403	2440			45	
	230	12	1500	355	54	108	65	16	258	2368	2440			45	
	AF-1	12	1500	317	*	*	148	34	*	2011	2440			45	
	AF-2	12	1500	203	*	*	77	28	*	1820	2440			45	

*Included with experiments

Note: For power distribution purposes, it was assumed that 10 percent of experiment, 50% of SCS, and 90% of data processing power was ac power.

For example, the experiment energy for Flight 215 was initially estimated to be 242 kilowatt-hours but was reduced to 174 kilowatt-hours by the computer scheduling.

The CRT readout, added to other constant and variable loads results in a power load profile typical of that shown in Figure 32. This particular portion of the profile was chosen because it represents one of the more demanding periods of time on this particular flight.

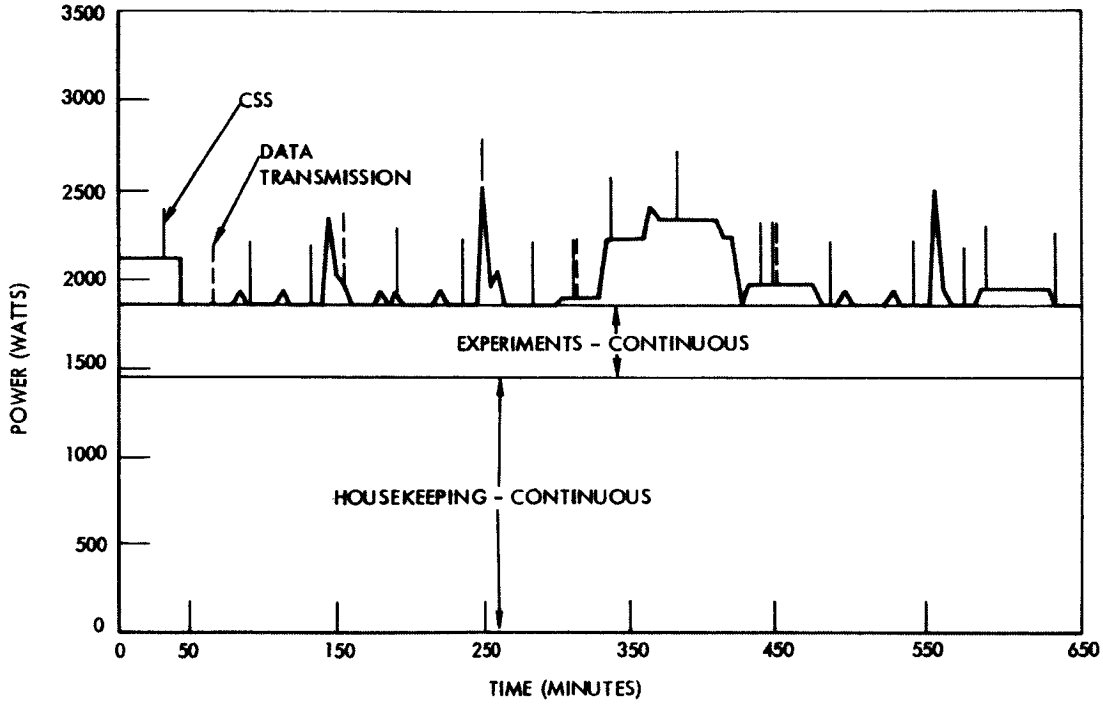

Figure 32. Integrated Power Load Profile, Flight 215 - Day 3

Subsequent to the power load vehicle capability analysis summarized in Table 32, computer runs were made on other flights and the average power consumption was determined. These results are shown as follows:

Flights	Average Load Charged to Experiments (Watts)
209	416
211	440
507	816
509	419
215	520
513	439
219	419
221	411
518	824*
229	650
230	424
AF-1	
AF-2	

*Solar cell experiment provides extra power on this flight.

DESIGN ALTERNATIVES

Within each of the basic configurations, there are many tradeoffs and design alternatives involved, particularly in the area of modifications to the command-service module versus duplication of equipment and functions in the external device.

Significant weight saving can be obtained in many cases by adding provisions that allow greater use of CSM equipment.

The energy requirements for Configuration 1 flights are such that up to 1800 pounds of batteries are required in the external device to achieve a 14-day mission. Alternatives to providing these batteries would be shortening the mission or modifying the CSM in such a way that additional reactants are provided to the fuel cells, either by adding tanks to the service module or piping reactants from the external device to the service module. These alternatives were considered during the study but were not adopted because of the amount of service module modification required. More detailed information on the modifications required and the feasibility of this approach may be found in the appendix.

Another major area of design alternatives concerns the use of CM instrumentation for power system components in the external device. This section defines the interface alternatives and describes the recommended umbilical connections.

INTERFACE BETWEEN CSM AND EXTERNAL DEVICE

Configurations D and D' require that a new energy source be installed in the external device. The variations on this approach include simply putting additional hydrogen and oxygen in the device and piping the reactants to the service module power system, or installing a separate power system complete with instrumentation, controls, inverters, power distribution, etc.

The Apollo Block II power distribution system is based on a 28-volt, two-wire d-c system and a 115/200-volt, 400-cps, 3 phase, four-wire ac system. Both systems are grounded to the spacecraft structure at a single point in the command module. All loads are grounded to this point through their respective negative or neutral bus.

The main dc bus system is comprised of isolated and redundant positive buses A and B, which receive their power from the three fuel-cell

modules located in the service module and/or the reentry batteries located in the command module. The buses are sized so that each power-consuming device is provided a minimum of 25 volts at maximum load, excluding transients or emergency conditions.

The main ac bus system consists of independent, redundant buses 1 and 2 that receive their power from one of three 1250-VA redundant static inverters in the command module. The inverters cannot be paralleled but each bus can be supplied from a separate inverter. Motor switches, powered from the dc bus, are used to connect the inverters to the buses. The ac buses are located in the command module; all ac loads in the service module are powered separately from the command module buses.

Operation, monitoring, and control for the fuel cells, cryogenic storage system, and fuel-cell heat rejection system is accomplished from the command module. A summary of the required functions and the wiring characteristics that are used on Apollo are shown in Table 33, which shows that approximately 168 wires are routed between the command and service modules for power distribution and power system instrumentation purposes. In order to use the same control and instrumentation components for a fuel-cell power system located in the external device, this system would have to be essentially duplicated and a means provided to change the inputs to each instrument from one power system to the other. Such a scheme would require extensive modification of the Apollo Block II CM wiring, which is not in accordance with the objectives of the AES program.

It should be noted that the wiring list shown in Table 33 is based on the Apollo case of three fuel cells and two sets of H_2 and O_2 tanks. Any variation in numbers of components must, of course, be reflected by corresponding changes in wiring and instrumentation. For example, in Configuration D' where more than two hydrogen and two oxygen tanks are required in the external device, additional instrumentation must be installed for the extra tanks.

The inverters are self-regulating and have little instrumentation other than for monitoring the characteristics of power on the ac buses. Therefore, from an instrumentation standpoint, the Apollo CSM control and display panel can be used to monitor the inverter output, regardless of inverter location. A more important criterion in this case is inverter reliability. Reliability analyses (explained later in this report) showed that addition of inverters in the external device in lieu of interconnecting the ac buses actually lowered the mission success reliability.

The degree of modification to an Apollo command module required to connect approximately 170 wires to the right location and install selector switches in the instrument panel (that would determine which system would

be controlled or monitored) appears to be more complicated and less reliable than the objectives of this program require. Because the external device has a habitable area, the duplication of power system instrumentation in this area allows a simple interface—essentially, the extension of two CM dc buses and two ac buses. Although the umbilical may consist of several bundles of wires for flexibility and redundancy, the connection points (which are the critical modification indicators rather than numbers of wires) are few. For these reasons, it is recommended that instrumentation and control for the power generation system be installed in the external device. Warning indicators and general data transmission can be used to notify the flight and ground crews of any abnormal conditions.

ELECTRICAL UMBILICAL

In the present Apollo Block II configuration, there are two redundant electrical plugs in the airlock separating the command module from the LEM. The present connectors are primarily for instrumentation purposes and would not be applicable for the AES/command module power interface. Presently, there is a proposal to supply power to the LEM vehicle in the range of 2300 watts for approximately one-half hour. If this is the case, a power umbilical interface would be required on Block II configurations and possibly could be utilized for the AES/command module power interface. However, no details are available on connector and pin sizes. Furthermore, it has been mentioned that out of the proposed 2300 watts, there will be no ac transmission, thereby, nullifying any presently adaptable or proposed AES/command module ac transmission in Block II.

To supply the power demands of the Apollo vehicle from the external device and, in turn, supply the external device ac requirements from the Apollo inverters, it will be necessary to revise the present Apollo/LEM umbilical. A minimum of thirty-two 12-gauge, and eight 16-gauge wire and pin connections will be necessary. Thirty-two wires will connect into main buses A and B in the right-hand aft equipment bay at designated main terminal board buses A and B. The remaining other eight wires will be connected to the terminal strips of ac buses 1 and 2.

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Table 33. Power System Wiring From SM to CM

INSTRUMENTATION			
Subsystem	Function	No. of Wires	Wire Size (AWG)
CSS	H_2 tank pressure	4	22
CSS	H_2 tank Signal conditioner	4	22
CSS	0_2 tank Signal conditioner	4	22
CSS	0_2 tank pressure	4	22
Power system Heat rejection	Temperature fuel cell No. 1 Radiator outlet	2	22
Power system Heat rejection	Temperature fuel cell No. 2	2	22
Power system Heat rejection	Temperature fuel cell No. 3	2	22
Fuel cells	Fuel cell No. 1	2	22
Fuel cells	Fuel cell No. 2 H_2 flow rate	2	22
Fuel cells	Fuel cell No. 3 H_2 flow rate	2	22
Fuel cells	Fuel cell No. 1 0_2 flow rate	2	22
Fuel cells	Fuel cell No. 2 0_2 flow rate	2	22
Fuel cells	Fuel cell No. 3 0_2 flow rate	2	22
Fuel cells	Temperature fuel cell No. 3 condenser exit	2	22
Fuel cells	Temperature Skin Fuel cell No. 3	2	22
Fuel cells	H_2 pressure regulator Fuel cell No. 3	2	22
Fuel cells	0_2 pressure regulator Fuel cell No. 3	2	22
Fuel cells	N_2 pressure regulator Fuel cell No. 3	2	22

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Table 33. Power System Wiring From SM to CM (Cont)

INSTRUMENTATION			
Subsystem	Function	No. of Wires	Wire Size (AWG)
Fuel cells	PH factor water Fuel cell No. 3	2	22
Fuel cells	Temperature fuel cell No. 2 Condenser exit	2	22
Fuel cells	Temperature Skin Fuel cell No. 2	2	22
Fuel cells	H_2 pressure regulator Fuel cell No. 2	2	22
Fuel cells	0_2 pressure regulator Fuel cell No. 2	2	22
Fuel cells	N_2 pressure regulator Fuel cell No. 2	2	22
Fuel cells	PH factor water Fuel cell No. 2	2	22
Fuel cells	Temperature fuel cell No. 1 condenser exit	2	22
Fuel cells	Temperature Skin Fuel cell No. 1	2	22
Fuel cells	H_2 pressure regulator Fuel cell No. 1	2	22
Fuel cells	0_2 pressure regulator Fuel cell No. 1	2	22
Fuel cells	N_2 pressure regulator Fuel cell No. 1	2	22
Fuel cells	PH factor water Fuel cell No. 1	2	22
Fuel cells	Fuel cell No. 1 bus A disconnect	1	22
Fuel cells	Fuel cell No. 2 bus A disconnect	1	22
Fuel cells	Fuel cell No. 3 bus A disconnect	1	22
Fuel cells	Fuel cell No. 1 bus B disconnect	1	22

Table 33. Power System Wiring From SM to CM (Cont)

Subsystem	Function	No. of Wires	Wire Size (AWG)
INSTRUMENTATION			
Fuel cells	Fuel cell No. 2 bus B disconnect	1	22
Fuel cells	Fuel cell No. 3 bus B disconnect	1	22
Total No. 22 AWG wires in twisted, shielded pairs Total No. 22 AWG wires in separately shielded bundle		70 6	
POWER			
Fuel cells	Fuel cell No. 2 main bus A Off	1	22
Fuel cells	Fuel cell No. 2 main bus B Off	1	22
Fuel cells	Fuel cell No. 2 main bus A disconnect On	1	22
Fuel cells	Fuel cell No. 2 main bus B disconnect On	1	22
Fuel cells	Fuel cell No. 2 Neutral	1	
Fuel cells	Fuel cell No. 2 ØA	1	20
Fuel cells	Fuel cell No. 2 ØB	1	20
Fuel cells	Fuel cell No. 2 ØC	1	20
Fuel cells	Fuel cell No. 2 H_2 close	1	22
Fuel cells	Fuel cell No. 2 H_2 open	1	22
Fuel cells	Fuel cell No. 2 0_2 open	1	22
Fuel cells	Fuel cell No. 2 0_2 close	1	22
Fuel cells	Fuel cell No. 2 H_2-0_2 indicator close	1	22
Fuel cells	Fuel cell No. 2 purge valve H_2	1	22
Fuel cells	Fuel cell No. 2 purge valve 0_2	1	22
Fuel cells	Fuel cell No. 1 main bus A Off	1	22

Table 33. Power System Wiring From SM to CM (Cont)

INSTRUMENTATION			
Subsystem	Function	No. of Wires	Wire Size (AWG)
Fuel cells	Fuel cell No. 1 main bus B off	1	22
Fuel cells	Fuel cell No. 1 main bus A disconnect On	1	22
Fuel cells	Fuel cell No. 1 main bus B disconnect On	1	22
Fuel cells	Fuel cell No. 1 neutral	1	22
Fuel cells	Fuel cell No. 1 ØA	1	20
Fuel cells	Fuel cell No. 1 ØB	1	20
Fuel cells	Fuel cell No. 1 ØC	1	20
Fuel cells	Fuel cell No. 1 H_2 close	1	22
Fuel cells	Fuel cell No. 1 H_2 open	1	22
Fuel cells	Fuel cell No. 1 0_2 open	1	22
Fuel cells	Fuel cell No. 1 0_2 close	1	22
Fuel cells	Fuel cell No. 1 H_2-0_2 indicator close	1	22
Fuel cells	Fuel cell No. 1 purge valve H_2	1	22
Fuel cells	Fuel cell No. 1 purge valve 0_2	1	22
Fuel cells	Fuel cell No. 3 main bus A Off	1	22
Fuel cells	Fuel cell No. 3 main bus B Off	1	22
Fuel cells	Fuel cell No. 3 main bus A disconnect Off	1	22
Fuel cells	Fuel cell No. 3 main bus B disconnect On	1	22
Fuel cells	Fuel cell No. 3 neutral	1	22
Fuel cells	Fuel cell No. 3 ØA	1	20
Fuel cells	Fuel cell No. 3 ØB	1	20
Fuel cells	Fuel cell No. 3 ØC	1	20
Fuel cells	Fuel cell No. 3 H_2 close	1	22

Table 33. Power System Wiring From SM to CM (Cont)

INSTRUMENTATION			
Subsystem	**Function**	**No. of Wires**	**Wire Size (AWG)**
Fuel cells	Fuel cell No. 3 H_2 open	1	22
Fuel cells	Fuel cell No. 3 0_2 open	1	22
Fuel cells	Fuel cell No. 3 0_2 close	1	22
Fuel cells	Fuel cell No. 3 H_2-0_2 close	1	22
Fuel cells	Fuel cell No. 3 purge valve H_2	1	22
Fuel cells	Fuel cell No. 3 purge valve 0_2	1	22
CSS	Fan motors System 1	3	20
CSS	Fan motors System 2	3	20
Heat Rejection	Glycol pump No. 1	3	20
Heat rejection	Glycol pump No. 2	3	20
Power distribution from SM to CM:	SM negative bus to CM negative	16	12
	SM bus A to CM bus A	8	12
	SM bus B to CM bus B	8	12
	28-volt main bus A&B operational instrumentation	2	16
	AC neutral	2	20
Total power lines:	No. 22 AWG wires	35	
	No. 20 AWG wires	23	
	No. 16 AWG wires	2	
	No. 12 AWG wires	32	
Total power and instrumentation lines:	No. 22 AWG wires	111	
	No. 20 AWG wires	23	
	No. 16 AWG wires	2	
	No. 12 AWG wires	32	

POWER GENERATION

This section considers the application of both batteries and Apollo-type P&W fuel cells to each configuration of interest. The fuel cells in the service module in Configurations 1, D, and D' were assumed to be qualified to 400 hours without in-space start capabilities. Fuel cells installed in the external device must have in-space capabilities start and preferably (from a weight standpoint), 1000 hours of life. Configuration C and the external device design of Configuration D are based on 1000-hour fuel cells, while the D' design is based on 400-hour units.

The Apollo fuel cell modules are complete powerplants in the sense that all regulation, control, and auxiliary equipment required for normal automatic operation is contained within the fuel cell system itself. All that needs to be provided are the reactants, a heat rejection radiator, ac power, and normal instrumentation. Thus, the installation of this system in an external device is quite feasible.

To take full advantage of the experience gained on Apollo and to minimize the amount of testing required, the system design and installation should resemble the Apollo as much as possible. During this initial feasibility study, five areas received special attention because of differences in requirements from the Apollo vehicle; the operation of two fuel cells in parallel rather than three, i.e., fuel cell efficiency; the supplying of peak loads beyond fuel cell capability; in-space start; the operation of a fuel cell in a device other than the SM in parallel with one in the service module; and the EPS radiators on an external device.

ELECTRICAL POWER SYSTEM HEAT REJECTION

The Apollo Block II spacecraft is designed to operate in a variety of space conditions during a normal lunar flight. AES flights sometimes require a different orbit inclination, and orbit heights varying from 200 nautical miles to synchronous. In addition, there is sometimes a requirement for attitude-hold for experiment purposes. The analysis was directed toward determining if these differences would require significant changes in the design or operation of the Apollo radiator, and if the Apollo radiator design would be suitable for the external device electrical power system (EPS).

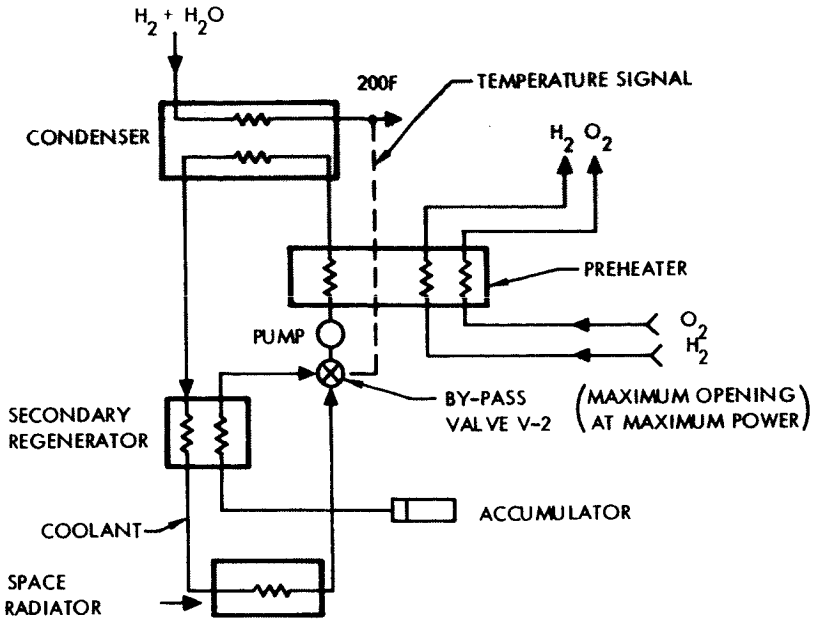

Figure 33. Fuel Cell Heat Rejection System

Figure 33 is a schematic of the Apollo EPS secondary heat rejection loop. Heat is rejected to the secondary coolant (62.5/37.5 ethylene glycol and water mixture) at the condenser thus maintaining the primary system heat balance. The condenser has a hydrogen exit temperature requirement of 195 ± 5 F. If the temperature is low, the sensor causes the secondary bypass valve to preheat through the regenerator. This regenerator must operate over a wide range, i.e., -35 to 190 F, in order to maintain a constant coolant temperature at the condenser. Any appreciable rise in temperature due to the bypass must be matched by a comparable drop on the hot side of the regenerator. Therefore, the radiator inlet temperature will vary with the action of the secondary bypass valve. This valve is designed to have maximum opening at maximum power. The reactant reheaters serve to heat the reactant gas from the supply system to a temperature acceptable to the fuel-cell stack. Even though the temperature rise of the reactant fuel is 200 F, the corresponding rise of coolant is less than 10 F due to the heat capacity of the coolant being greater than that of reactant flow rate. The requirement of the preheater is that the coolant inlet temperature be between 125 to 185 F. With this in mind it is possible to set the range of radiator exit temperatures which relate to given inlet temperatures. For example, at a radiator inlet temperature of 190 F, the exit temperature will be >90 F (assuming a maximum coolant temperature of 225 F). Below 90 F the required ΔT across the hot side of the regenerator would lower the radiator inlet temperature to below 190 F. The radiator temperatures

of inlet and exit conditions are shown in Table 34 illustrating typical fuel cell operation.

The glycol/water mixture is circulated through the secondary coolant loop by the action of a glycol pump. To show maximum and minimum radiator areas, flow rates were taken from 100 — 20 lb/hr. The pump circulates fluid at a nominal rate of approximately 45 lb/hr against system pressure losses (about 6 psi including the radiator). A prototype test operation of the secondary loop heat exchanger is shown by Figure 34.

The Apollo EPS radiator panel consists of panels of fixed area of approximately 5 square feet each. Each tube on a panel corresponds to one fuel cell; i.e., one parallel passage per fuel cell. Figure 35 shows a typical radiator section of 8 panels. The weight will be approximately 0.68 lb/sq ft plus any honeycomb background structure.

Figure 36 indicates maximum waste heat to be rejected as a function of gross power. These values represent maximum heat rejection at the interface (i.e., transfixed to the glycol in the condenser). Of this sum it is expected that possibly as high as 10 percent may be lost in the fuel cell heat rejection system; e.g., fuel preheater, secondary loop regenerator, condenser, pipe losses, etc. Also, more recent information leads to the conclusion that present design will permit lower heat rejection requirements due to improved efficiency. However, a conservative approach is shown to represent worst cases. The efficiency considered is calculated from a simple heat balance as follows:

$$Q \text{ rejected} + \text{gross power} = \text{total heat}$$

Or

$$\frac{P_{GP}(1-\eta_{OA})}{\eta_{OA}} = Q \text{ rejected}$$

Where

ϵ = emissivity

α = absorptivity

P_{GP} = gross power output

η = efficiency

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Table 34. Summary of Heat-Rejector System

Radiator Temperature (F)			Heat Load			
Inlet*	Exit** (Range)	Pre-Heater	Gross Power output per module	Overall*** efficiency η_{OA} (min)	Heat Rejected (max) Q_r Btu/hr	Effective Area A^1 (sq ft) (Range)
190	90 to 185	125	2.0	0.54	5900	29-32
160	60 to 155	125	1.6	0.60	3700	20-25
115	15 to 110	125	1.2	0.65	2200	15-19
65	-35 to 60	125	0.8	0.67	1400	14-20
65	-35 to 60	125	0.56	0.65	1000	10-14
5	-35 to 0	185▲▲	0.56	0.65	1000	16-18

Radiator Area (sq ft)/fuel cell Flow range (100-20 lb/hr)			
Nominal Qin=45 Btu hr-sq-ft (Range)	Nominal Qin=45 Btu hr-sq ft (Range)	$Peak_1$ Qin=80 Btu hr-sq ft (Range)	$Peak_2$ Qin=90 Btu hr-sq ft (Range)
34 - 39	38 - 43	47 - 57	52 - 59
24 - 30	26 - 36	35 - 51	38 - 85
18 - 25	22 - 31	33 - 62	39 - 85
18 - 32	25 - 50	~90▲	-
13 - 23	19 - 40	~65	-
32 - 37	64 - 84	-	-

*Based on 225 F (max) condenser exit

***-35° taken to be the lower limit

*** $OA = \frac{Q_r + 1}{P_{(gross)}}$

▲Reject temperature approaches effective heat-sink temperature corresponding to Q_{in}

▲▲Represent maximum pre-heater temperature

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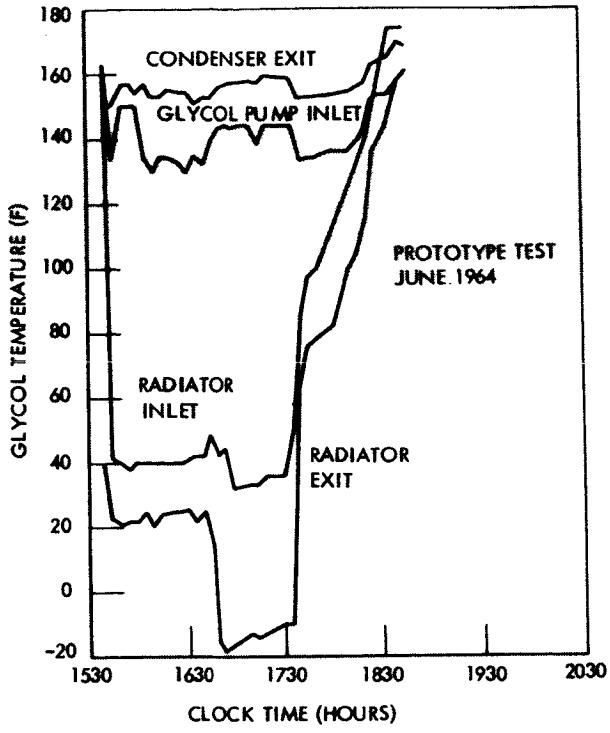
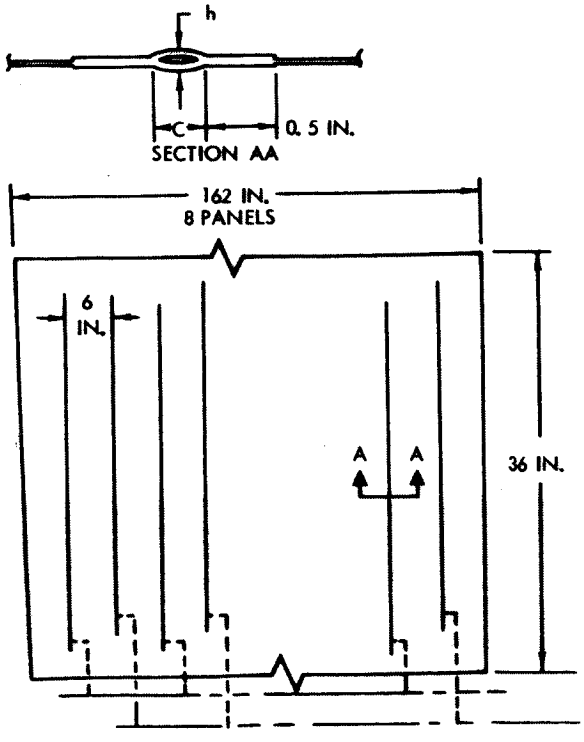


Figure 34. Secondary-Loop-Heat Exchange Operating Parameters in Apollo



AREA PER PANEL	5 FT^2 (36 IN. X 20 1/4 IN.)
FIN THICKNESS	0.032 IN.
TUBE GEOMETRY SPACING	CIRCULAR SEGMENT h = 0.100 IN.; C = 0.625 IN.; 6.0 IN. CENTERLINE TO CENTERLINE
COOLANT TYPE FLOW RATE	62.5 PERCENT (WEIGHT) AQUEOUS ETHYLENE GLYCOL 20 TO 90 POUNDS PER HOUR
SURFACE CHARACTERISTICS	SOLAR ABSORPTIVITY = 0.27-0.15 INFRARED EMISSIVITY = 0.90-0.85

Figure 35. Electrical Power System Radiator Panel

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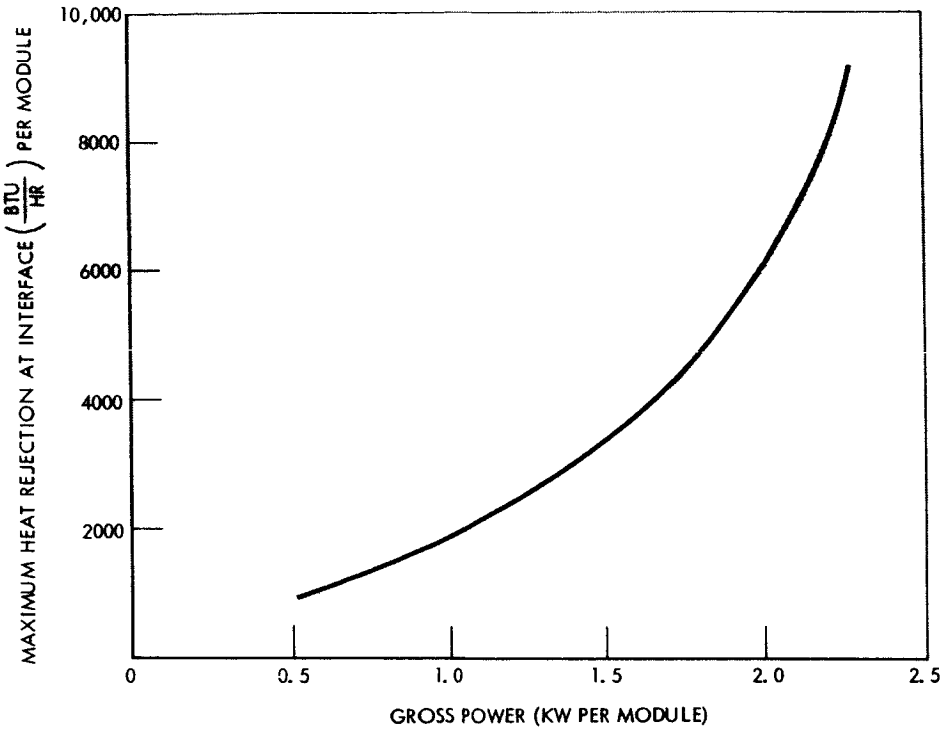


Figure 36. EPS Radiator Heat Rejection Specification

A^1 = effective area

Ω = fin effectivenss

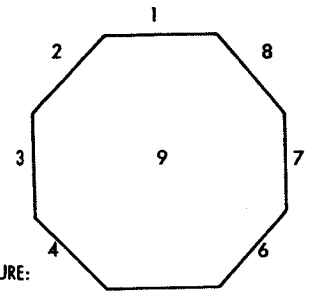
A = area

T_s = effective heat-sink temperature

T_h = radiating temperature

The orbital thermal environment is very critical to the determination of actual radiator area requirements. For the Apollo Block II radiator the poorest fin effectiveness is about 0.80. However, if random heat inputs are considered for this mission, fin effectiveness values vary to extremes and areas required become excessive. An example of this is presented by considering a case of a low altitude, low inclination orbit. If the spacecraft is assumed to have nine flat sides as shown in Figure 37, and is Earth-oriented with surface 9 (Figure 37) pointing to the earth, heat inputs for each of the nine sides will be the values shown in the figure. This, however, is based on a stationary spacecraft and static conditions. Because of the rapidly changing environmental heat loads during a normal orbit, a transient analysis of the complete system should be performed to provide more detailed data.

SIDE	1	2	3	4	5	6	7	8	9
INPUT $\frac{BTU}{HR}$									
MAXIMUM	30	80	105	80	30	80	105	80	85
AVERAGE	25				25				
MINIMUM	19	19	19	19	19	18	18	18	55



LOW ALTITUDE, LOW INCLINATION (SIDE 9 POINTING TO EARTH)

EFFECTIVE HEAT SINK TEMPERATURE:

$Q_{in.} = \sigma \epsilon'_T T_s^4$

$T_s = \left(\frac{Q_{in}}{\sigma \epsilon_T}\right)^{1/4}$

Figure 37. Heat Input Parameters for EPS Radiator

Calculations for various radiator inlet conditions and nominal values of heat input as shown in Figure 37 resulted in the radiator design parameters in Table 34. At full load, the fuel-cell heat-rejection temperature and, therefore, the radiator inlet temperature will be in the 160-190 F range. This results in a nominal radiator size of 20 to 30 sq ft per fuel cell or 40 to 60 sq ft total for the Configuration D and D' external device. It is emphasized, however, that a mission requirement of holding the radiator surface toward both solar and earth reflection heat inputs for an extended period at full power load would result in radiator overheating. Therefore, mission requirements causing the vehicle to be held at a particular attitude for long periods should be examined considering radiator characteristics and location before the radiator size and design become fixed.

FUEL-CELL POWER SYSTEM OPERATION

AES Configurations D and D' have two fuel-cell power systems and two cryogenic storage systems. Ideally, the service module system supplies all loads for the first 14 days and the external system supplies all loads thereafter. However, as may be seen in the discussion on the cryogenic storage system, situations may arise when both systems should be operating at the same time in order to use up all reactants in the service module and still not waste hydrogen from boiloff in the external device. Also a situation could possibly arise where two of the service module fuel cells would fail prematurely and the third one would continue to be used until all service module reactants were consumed.

Operating One Service Module Fuel Cell in Parallel with One External Device Fuel Cell

In the absence of a hydrogen interconnection between the external device and service module, at least one fuel cell module from one system would operate simultaneously with at least one module from the other. In such a case, there are at least three alternatives available: (1) allowing the fuel cells to share the load in whatever way the conditions of the fuel cells and system dictate; (2) forced load sharing by placing one fuel cell on bus A and another on bus B, and manually switching the loads to bus A or B as required; and (3) modifying the external device fuel cells so that their operating temperature can be manually controlled, thus controlling their voltage and load-sharing characteristics.

Operating the Two Systems on Separate Buses

The Block II Apollo spacecraft has redundant buses. Selection of the bus to use for a particular load is either manually controlled by the crewman or by connecting the load to each bus by diodes to allow current to pass from either dc bus but not from one dc bus to another. Much of the load (including all ac loads) is manually controlled, thus making it feasible to force fuel-cell load-sharing in this manner. However, for reliability reasons, this method should not be used in a normal mission, since if the mission success depends on such a scheme, the dc buses can no longer be considered redundant.

Normal Load Sharing Between SM Systems and External Device

Two identical fuel cell powerplants located side by side, with identical transmission lines to common positive and negative buses, can share loads equally. If transmission lines are unequal, different amounts of power are lost in the transmission lines, causing unequal load-sharing. Effects of transmission line upon deliverable or bus power were computed over the complete range of power generation.

The net effect was found to be very slight for the small differences in cable length between the SM and the external device and their common load. Therefore, if the fuel cell temperature and other characteristics in the two systems are alike, load sharing will be approximately equal between them. Figure 38 shows the voltage characteristics resulting from operating temperature and also line losses. As the figure shows, operating temperature has by far the greatest effect.

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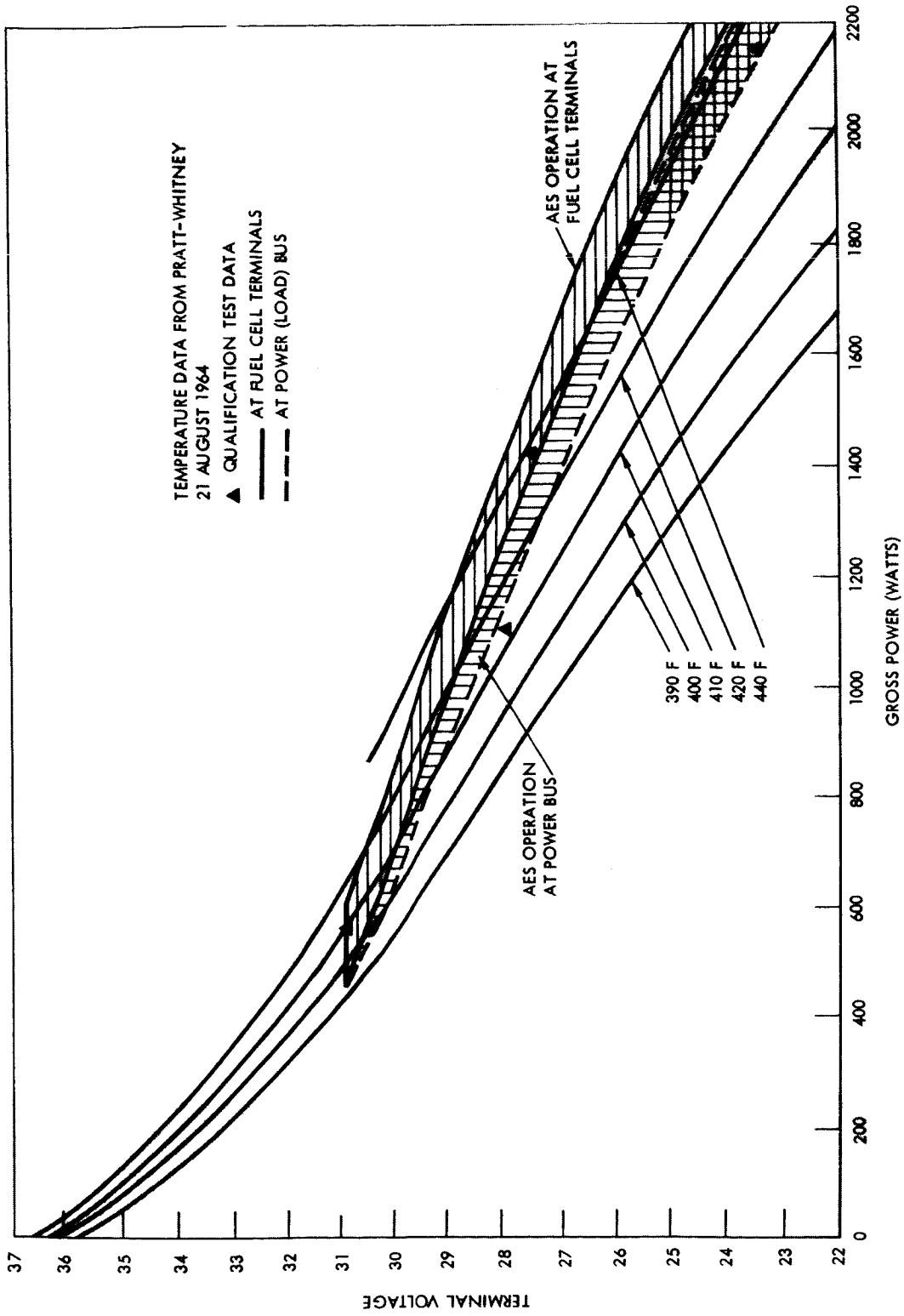

Figure 38. Effect of Operating Temperature on Fuel Cell Voltage

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Load-Sharing Controlled by Dual or Multiposition Primary Bypass Valves

Service module (SM) and AES External Device fuel cell powerplants are controlled through primary and secondary regenerator bypass regulator valves, both of which are in turn thermomechanically controlled. The secondary bypass regulator attempts to control the fuel cell along a preselected constant-temperature voltage — gross-power characteristic function. The primary bypass regulator permits a programmed variation of function. As indicated in Figure 38, the primary bypass regulator permits a programmed variation of temperature along the voltage-gross-power characteristic. The valve barely opens at the minimum power of 653 watts and at the preselected temperature; it is completely open at maximum power of 1420 watts, with a cell-stack temperature approximately 40 F higher than the preselected minimum power temperature. Valve position between the power settings is intermediate, resulting in cell stack temperatures as indicated in Figures 38 and 39. Figure 39 was constructed from Pratt & Whitney data by plotting voltage-power characteristics with various primary regulator settings. Constant temperature characteristics are included. Primary bypass valve operation, as described, results in an improved (more horizontal) power characteristic as indicated in Figure 38. If desired, these valves can be modified to include two-position control.

Maximum variation in load-sharing between two fuel cells can be obtained by operating one at maximum, the other at minimum power. From Figure 39 it is seen that if one fuel-cell primary regulator setting at minimum power (barely open) corresponds to cell-stack temperature between 350 and 360F, and the other fuel cell primary regulator setting at minimum power is between 410 and 430 F, maximum variation defined in this manner is possible. A valve set for minimum power with a cell-stack temperature of 350 F, opens completely at the maximum power condition, with the cell stack temperature at 390 F.

From Figure 39, power characteristics corresponding to extreme variations in valve settings were constructed, resulting in Figures 40 and 41. The normal power characteristic was also included in both figures. With two fuel cells sharing the load, a load-sharing chart (Figure 42) was constructed from Figures 40 and 41. Typical variation due to minor variations in fuel cell construction, were included. Figure 43 gives power characteristics for normal, lowest, and highest temperature settings.

In this study, the service module fuel cell was not considered modified, therefore its primary bypass regulator has the normal setting. If while the service module and AES external device fuel cells are sharing load equally, and for some reason the crewman desires to increase the service module fuel-cell load, he proceeds to the external control panel and moves the primary bypass valve to the low temperature setting. The S/M

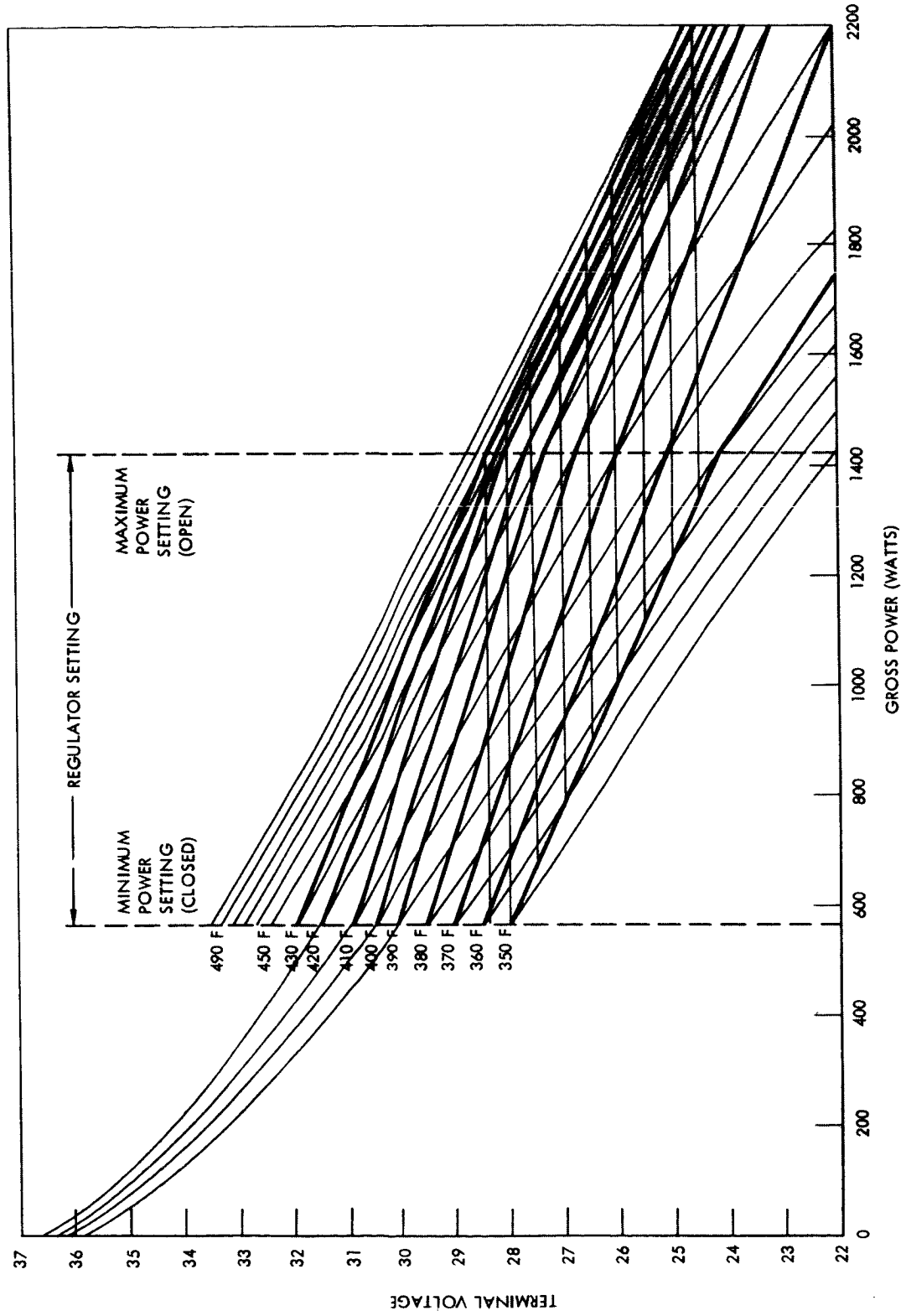


Figure 39. Effect of Primary Bypass Regulator Setting on Fuel Cell Operation

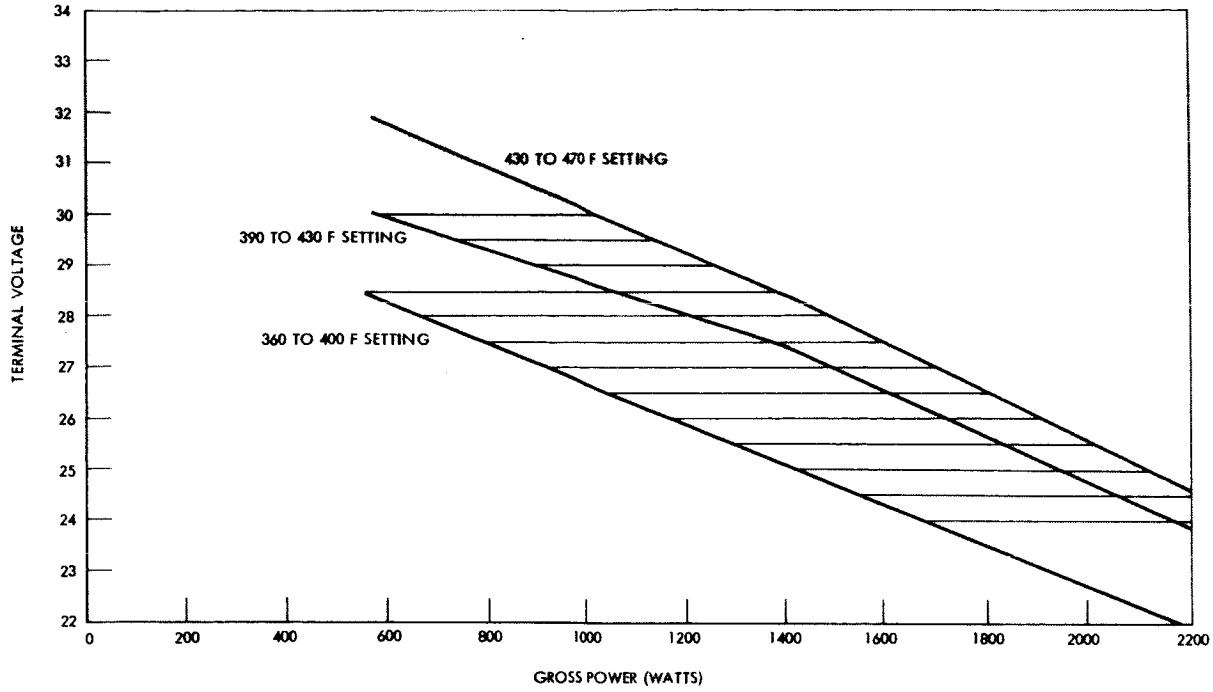


Figure 40. Fuel Cell Unbalanced Load-Sharing Controlled by Primary Bypass Two-Position Regulator with Minimum Power Settings of 360 and 430 F

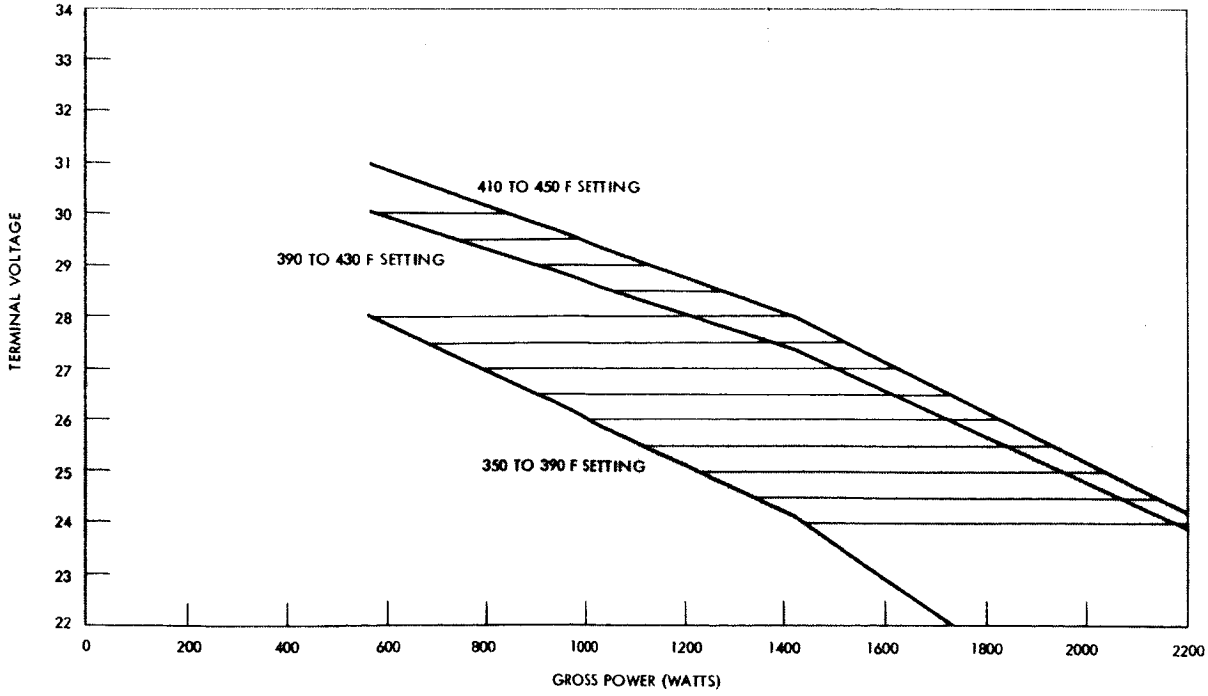


Figure 41. Fuel Cell Unbalanced Load-Sharing Controlled by Primary Bypass Two-Position Regulators with Minimum Power Settings of 350 and 418 F

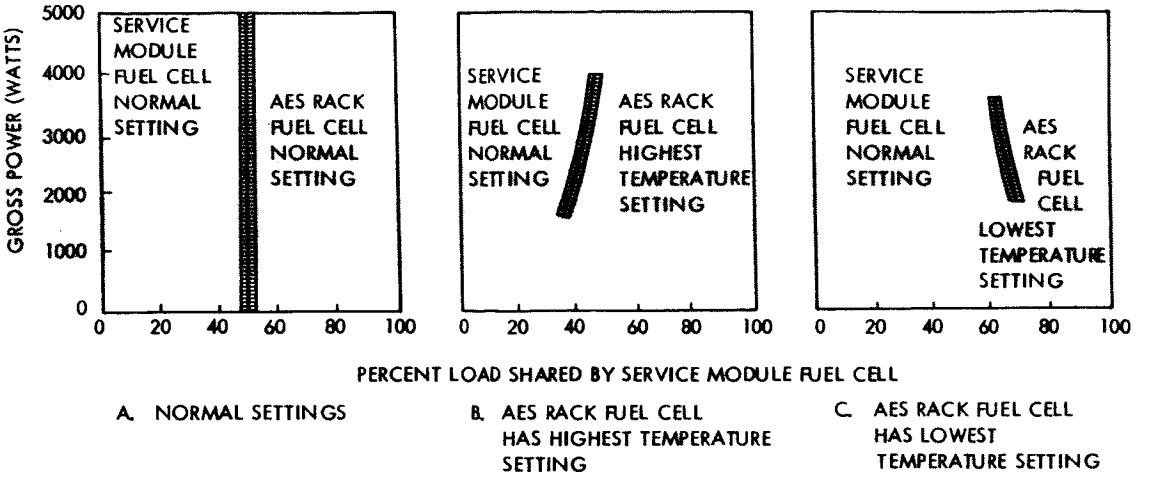

GROSS POWER (WATTS)
5000
4000
3000
2000
1000
0
SERVICE MODULE FUEL CELL NORMAL SETTING
AES RACK FUEL CELL NORMAL SETTING
SERVICE MODULE FUEL CELL NORMAL SETTING
AES RACK FUEL CELL HIGHEST TEMPERATURE SETTING
SERVICE MODULE FUEL CELL NORMAL SETTING
AES RACK FUEL CELL LOWEST TEMPERATURE SETTING
0 20 40 60 80 100
PERCENT LOAD SHARED BY SERVICE MODULE FUEL CELL
A. NORMAL SETTINGS
B. AES RACK FUEL CELL HAS HIGHEST TEMPERATURE SETTING
C. AES RACK FUEL CELL HAS LOWEST TEMPERATURE SETTING

Figure 42. Percentage Load-Sharing of Service Module Fuel Cell with AES Rack Fuel Cell

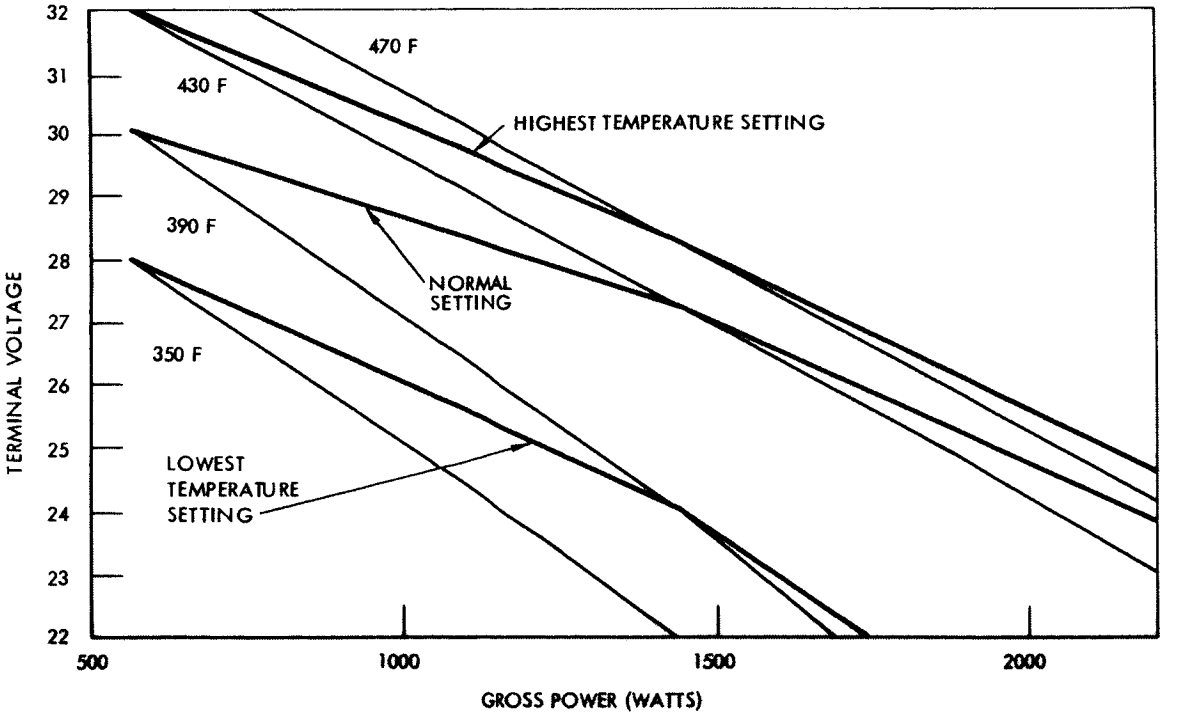


Figure 43. Power Characteristics of Apollo and AES Fuel Cells With Various Settings of Primary Bypass Regulator

fuel cell then accepts a greater portion of load, according to Figure 42. If the service module fuel cell is required to share less than half the load, the astronaut moves the AES primary bypass valve to the high temperature setting. The service module fuel cell load then shifts to lower values (Figure 42). Regardless of whether the AES valve setting is one the high temperature or low temperature setting, load shared by each fuel cell tends to become equal at high power loads (low terminal voltage). By so doing, the astronaut can control (to some degree) the use rate of cryogenics at both SM and external device locations.

SUPPLEMENTARY BATTERIES FOR PEAK POWER LOADS

The Apollo spacecraft uses the silver-zinc entry batteries not only for power when the command module is separated from the service module, but also to supplement the fuel cells whenever the power demand exceeds the capability of the fuel cells. During a normal lunar mission, this occurs only during checkout or firing of the service propulsion system (SPS). The number of charge-discharge cycles these batteries get from this operation are well within the capability. For the AES mission, however, it is inadvisable to use these batteries for all peak loads if the AES peaks (caused by experiments) are more numerous and severe than those for a lunar mission. The longer 30- or 45-day AES missions also tend to degrade the reliability of the entry batteries and this, added to the effect of increased charge-discharge cycles, might compromise the availability of energy for entry and post-landing requirements. It is therefore recommended that additional batteries be installed in either the service module or the external device to provide for peak loads.

At the time of the writing of this report, the mission experiments were not scheduled and so it was not possible to determine the magnitude and duration of each peak for each mission. Therefore, the sizing of supplementary batteries was done parametrically, and no attempt was made to arrive at a particular weight and volume for a particular mission. This weight is expected to range from zero where the requirements are similar to Apollo, to 380 pounds on Flight 518 to provide capability for 10 minutes of radar operation during Earth-shadow operation.

Parametric studies were made using silver-cadmium secondary batteries for sustained peak loads up to 10 kw and requiring up to 3 kwh of energy. The results of these studies are summarized in Figure 44 which can be used to determine the approximate supplementary power system characteristics for a wide range of required capabilities. Note that both peak power and total energy in the peak must be checked with the figure to see which is the critical parameter in determining battery size.

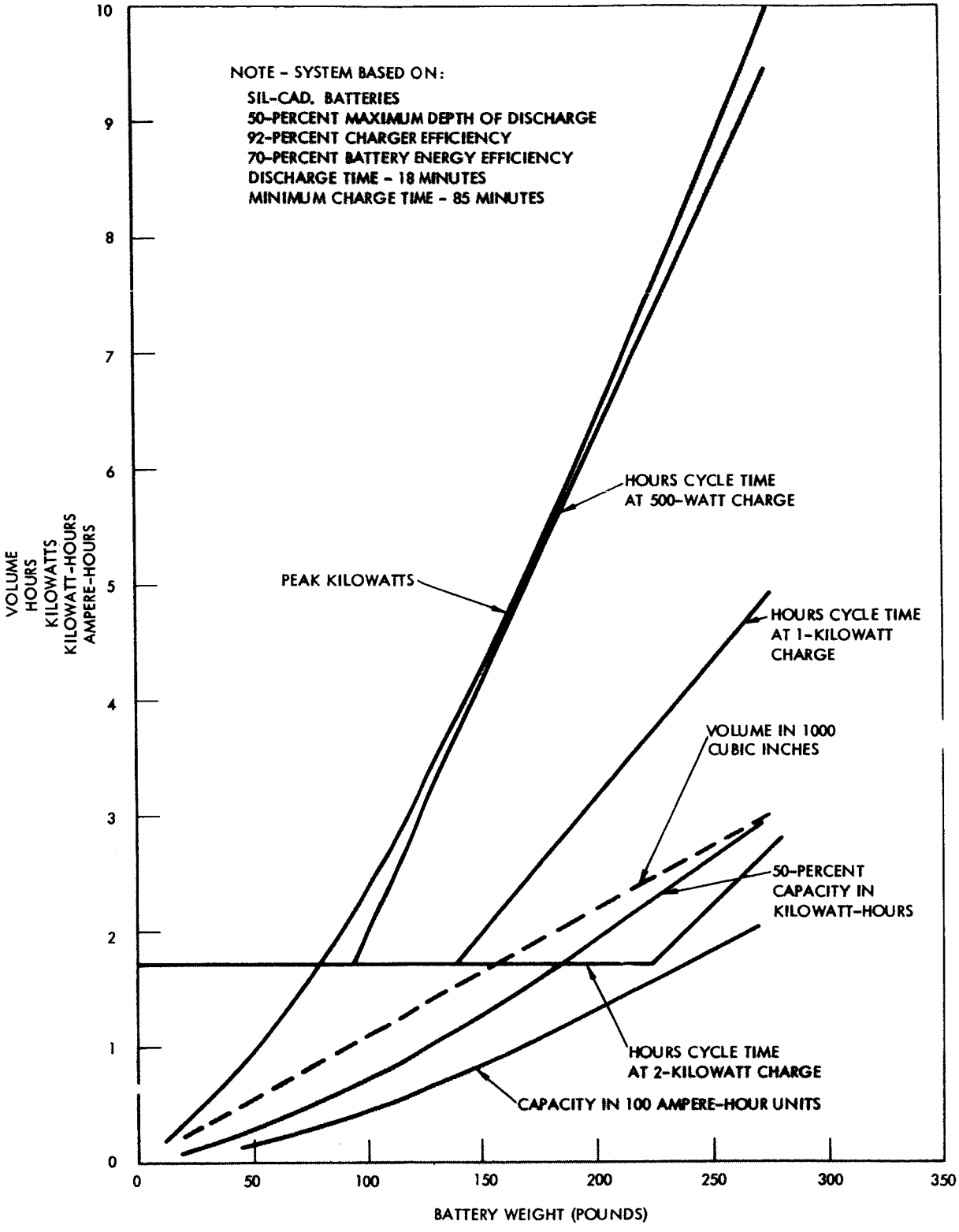


Figure 44. Battery Characteristics as Function of Total Battery Weight

If power peaks are frequent and/or unpredictable, an automatic system such as shown in Figure 45 is probably warranted instead of the Apollo type manual system. Because the charge-discharge characteristic of silver cadmium batteries differs from the silver zinc batteries used for energy storage in the CM, a new battery charger has to be designed. A current-metering device should be installed in the fuel cell power line providing a charge-limiting signal to the battery charger when any fuel cell reaches its full output. This would automatically restrict the battery charge cycles to the periods when other loads are below peak fuel-cell power loads. The battery charger design will not be seriously complicated by incorporating an output limiting device and will substantially simplify the overall system design.

A current-metering device at the fuel cell can activate the switch connecting the batteries on the dc bus line, but this would entail the disadvantage of repeated unnecessary switching. Another and simple method of connecting the batteries to the dc bus is using a dc undervoltage sensor which activates the battery switch. This undervoltage sensor could be activated at 26.5 volts ± 0.5 volts and would connect the batteries to the dc bus.

When voltage returns to a level of 31 ± 0.5, the batteries should be automatically disconnected from the line. At this voltage level, the total load will be equal or less than 2840 watts and can be handled by two fuel cells. The load distribution between fuel cells and battery will vary considerably according to fuel-cell temperature, and voltage sensing will not provide sufficient indication for disconnecting the battery. A current-metering device in each d-c bus will be necessary to provide a reliable signal for disconnecting the battery. This current-metering signal has to be based on the maximum power rating of two fuel cells. The approximate setting would be 90 ± 3 amp. When the current drops below this level, the batteries will be disconnected. When the batteries are discharged beyond a certain recommended state, a warning light will appear on the main control panel but they will not be removed automatically from the line.

Reverse current protection at the battery output under normal operating conditions is not needed since the normal battery charge is substantially above the discharge voltage. The reverse-current protection feature of a switch assembly would be used only if several battery cells would be shorted and the battery became unusable. In this case, the battery charger will also be disconnected from the line.

As described above, the added system would not change the Apollo subsystems characteristics of spacecraft power. The use of the silver zinc batteries located in the CM would be restricted, however, to reentry use, since all possible peak loads would automatically be powered from the silver cadmium batteries.

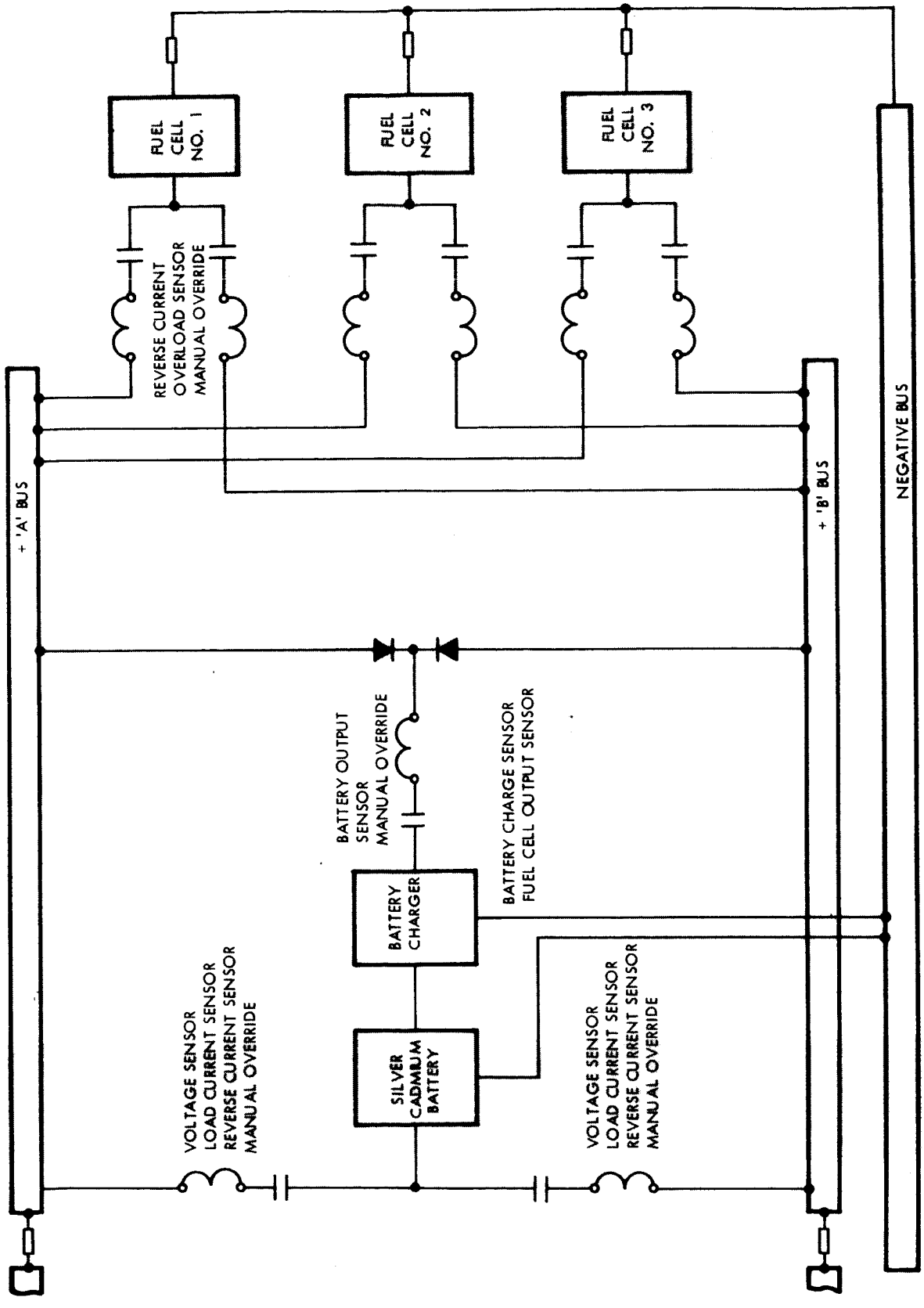


Figure 45. Simplified Schematic Diagram for Apollo Supplementary Power System

Approximate load sharing between the batteries and fuel cells can be determined from Figure 46. Test data indicates that the battery voltage will assume the line voltage in less than 2 milliseconds after paralleling and will not trip the reverse current protection of the fuel cells. No reverse current will occur in either of the fuel cells. A voltage transient will occur on the dc line any time the battery power is switched on or off. This transient, however, will not exceed the steady state voltage limits permissible for the power system.

IN-SPACE STARTUP UTILIZING BATTERIES

In-space startup of Pratt & Whitney (P&W) fuel cells may be accomplished by several means, but the method requiring the least development employs alkaline space batteries. The feasibility of this method has been assured by recent NASA/P&W efforts. With this method of startup, the specified 60-vdc heaters may be utilized by sizing the batteries to the following-desired requirements:

Voltage (dc)	Temperature (F)	Energy (kwh)
60	0 to 300	2.8
60	0 to 400	3.5

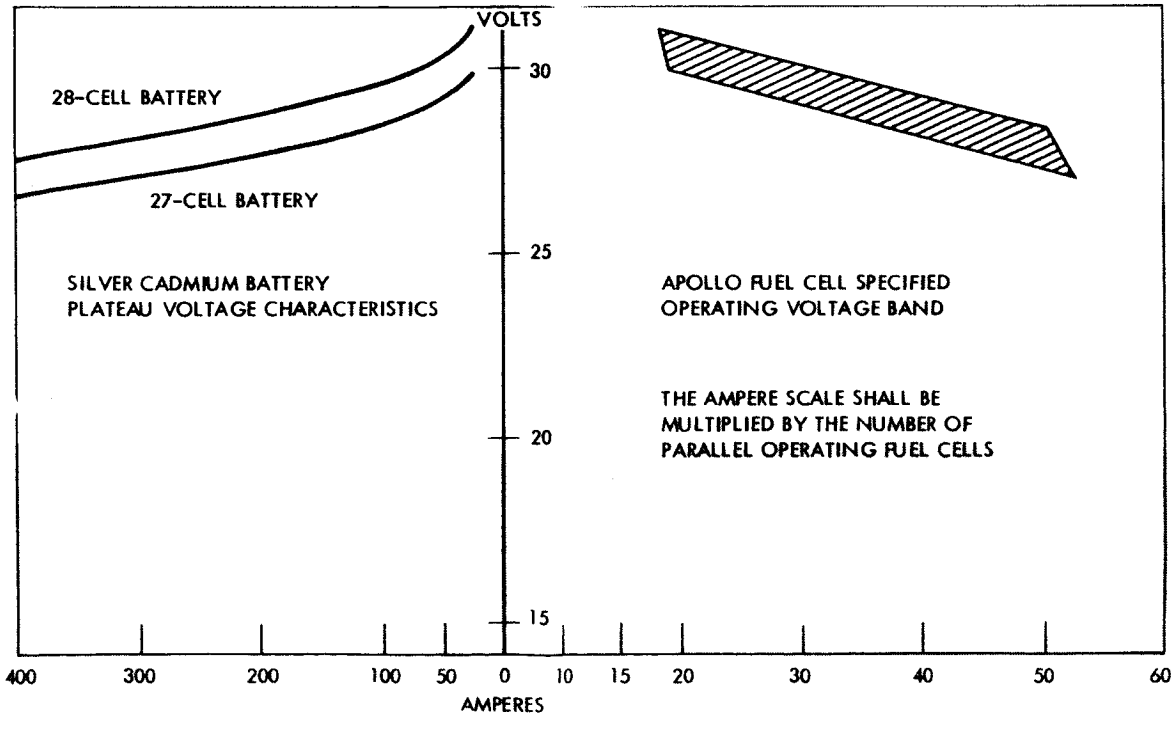


Figure 46. Apollo Supplementary Power System Voltage Characteristics

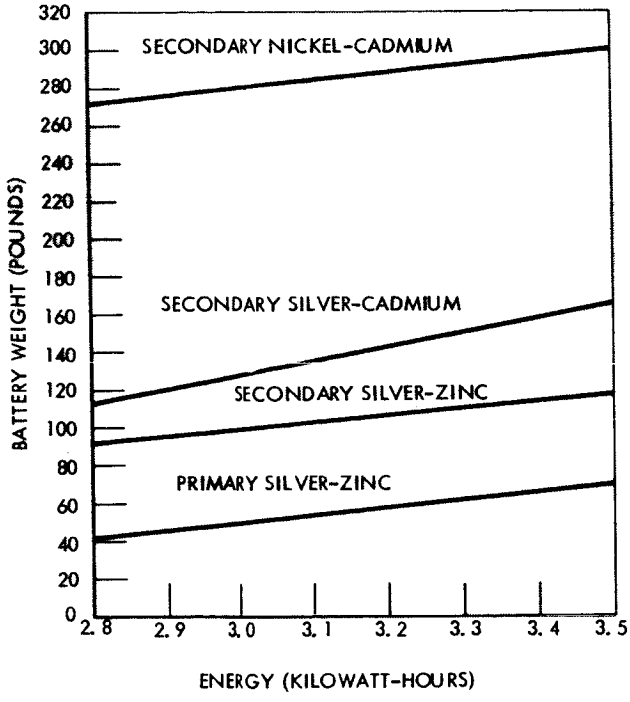


Figure 47. Required In-Space Start-Up Battery Weights

Figure 47 shows the weight differentials between the secondary batteries and a primary silver-zinc battery. The primary battery consists of 40 cells and is capable of restarting only one fuel-cell module. Comparable weight penalties are required for additional modules. The secondary silver-zinc battery is capable of 8 to 10 cycles and is composed of 48 silver-zinc cells. The silver-cadmium battery contains 60 such cells and is capable of several hundred cycles within representative weights. The nickel-cadmium battery contains 50 cells and will deliver several thousand cycles within specified weight limits.

When starting up only one or two fuel-cell modules, it is advantageous to use a primary silver-zinc battery. For more than two, a secondary silver-zinc battery with a battery charger is preferable. The fuel cells can also be started from operating fuel cells, but it is recommended that this be considered the backup mode, since the low system voltage (28 vs 60) and low amount of power remaining from spacecraft requirements cause the startup time to amount to several hours rather than approximately one hour.

FUEL-CELL EFFICIENCY

The efficiency of Apollo fuel cells depends upon several factors, including operating temperature, condition of the electrodes, internal losses, and electrode current density. In order to convert spacecraft electrical

requirements into reactant quantities, the efficiency or specific propellant consumption (pounds of reactants per kilowatt-hour of electricity) must be taken into account. During this study, the SPC was assumed to be the same as the average SPC over the Apollo lunar mission. However, since fuel-cell efficiency changes with load, the total electrical energy available when operated to one load profile is not necessarily the same as that available when operated to a different profile. The available electrical energy estimates were made on the assumption that the average operating efficiency would be the same as for Apollo. Flight 215 (the only flight where a complete power load profile was available) was used as a check to determine the validity of this assumption. The computer CRT profile (based on computer scheduling of the experiments) plus continuous experiment loads, continuous housekeeping loads, and variable housekeeping loads were analyzed to determine the percentage of mission time the fuel cells operated at various power levels, and the hydrogen consumption at each level. The results are shown in Table 35.

Table 35. Fuel Cell H_2 Consumption Analysis

Power Level (watts)	% of time at the Level	Fuel Cell H_2 Efficiency (lb/KWH)
1840	51.50	0.0831
1940	4.13	0.0804
2040	1.48	0.0796
2190	39.60	0.0787
2340	2.69	0.0762
2840	0.30	0.080
>2840	0.30	0.0915*

*Plus battery efficiency of 70%

With high pressure loading, each Block II H_2 tank contains about 28 lb of usable hydrogen. Purging the fuel cells takes about 0.00432 lb of H_2 per kwh of output (based on hydrogen purity of 99.94 percent); thus about 3 lb are required for this purpose. An allowance was made for fuel cell efficiency degradation as the mission progresses, 10 percent at the end of the mission— 5 percent average. It appears that the total load will never be

below the fuel-cell self-sustaining threshold of 563 watts per cell, and so no allowance was made for fuel-cell heaters.

Based on these analyses, the use of three fuel cells to supply the load profile (on a percentage basis) for Flight 215 would produce approximately 623 kwh of electrical energy. Since the available energy using Apollo efficiencies was 620 kwh, this verifies the initial thesis that the fuel cell efficiency does not vary significantly for any load profile in which most of the loads are relatively constant and the actual load profile is usually within a few hundred watts of the average load. This is provided, of course, that the total load does not go below the self sustaining threshold. Therefore, the use of 620 kwh as the energy contribution of the SM in Configurations 1, D, and D' seems to be reasonable.

RELIABILITY

The power system reliability analysis during the study was limited to fuel cells and inverters; other power system items were covered previously in the Apollo X study for mission durations up to 45 days. The primary purpose of the reliability analysis was to determine the number and type of fuel cells required for Configurations C, D and D', and the number and location of inverters.

FUEL CELLS

The fuel cell analysis was performed with both the 400-hour and 1000-hour fuel cells for different levels of confidence for each type of cell, and for different logical situations. Based on recent P&W test results and NASA concurrence, 1000-hour fuel cells were assumed to be available for all Configuration C and D flights, but not for the Configuration D' or 1 flights. Apollo X P&W failure-rate information and the Apollo X reliability goal of 0.995 with a confidence level goal of 90 percent was used for all analyses. These goals were not considered absolute criteria: neither the goals nor the calculation methods are sufficiently accurate for this.

The following ground rules were utilized in performing the analysis for Configurations D and D'.

1. In the SM, all 3 cells are started prior to launch
2. The cells in the SM may only operate for a maximum of 14 days.
3. If 2 cells in the SM fail, the additional cell in the SM will be turned off, and 2 cells in the rack will be activated.

Because of the situations dependent upon this logic, Markov chains were utilized to develop the reliability model. A typical state-space diagram used to compute the model for the situation of 3 cells in the SM and 3 additional cells in the external device (a logic where 2 cells are required at all times) is shown in Figure 48. Let A, B, and C refer to the fuel cells in the SM and D, E, and F refer to the cells in the external device.

The state-space diagram contains all the possible acceptable states of success for the first 14 days of the mission. From this diagram a

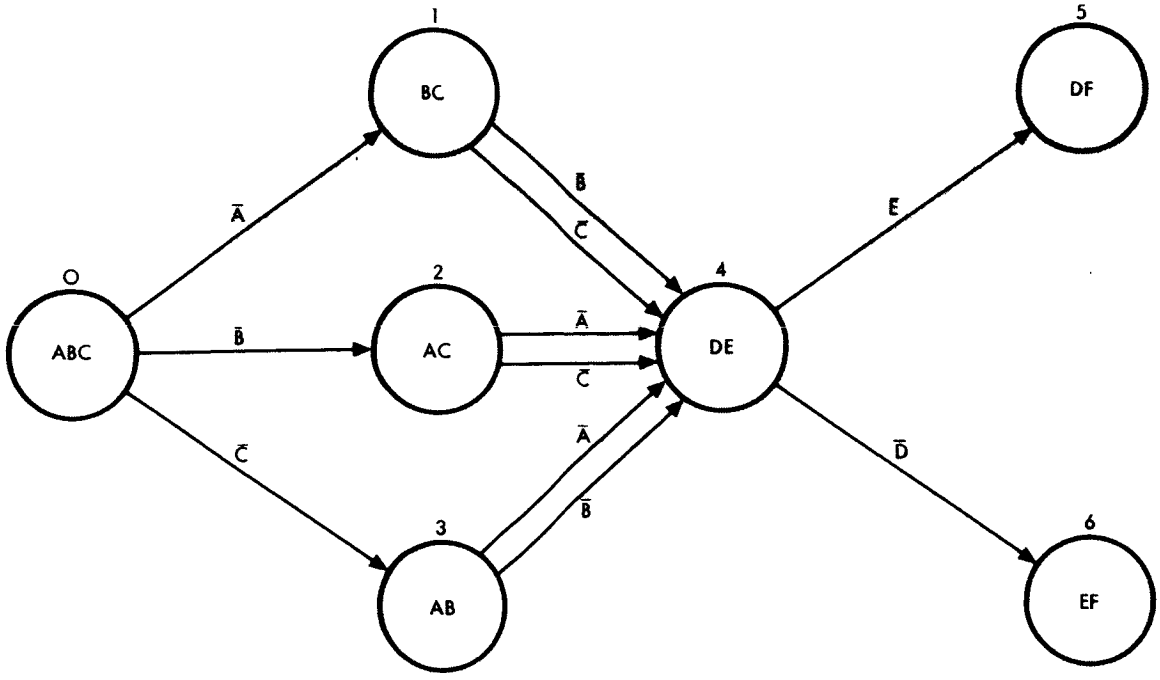


Figure 48. Reliability Space-State Diagram

transition matrix is determined, differential equations are formed, Laplace transforms are taken, inverse Laplace transforms are taken, and the final model is obtained which for this specific case is

$$R_{14\ \text{days}} = -8e^{-3\tau t_1} + 9e^{-2\tau t_1} - 6\tau t_1 e^{-2\tau t_1} + 6\tau^2 t_1^2 e^{-2\tau t}.$$

A similar approach determines the model for the remainder of the mission; for example, the next 16 days:

$$R_{16\ \text{days}} = e^{-2\tau t_2} + 2\tau t_2 e^{-2\tau t_2}.$$

The complete model for the specific situation would be given below

$$R_{30\ \text{days}} = (-8e^{-3\tau t_1} + 9e^{-2\tau t_1} - 6\tau t_1 e^{-2\tau t_1} + 6\tau^2 t_1^2 e^{-2 t_1})$$

$$(e^{-2\tau t_2} + 2\tau t_2 e^{-2\tau t_2})$$

Models for all other configurations and contingencies were developed in a similar manner.

Configuration 1

The reliability of the basic Apollo Block II power system is inherently adequate for the 14-day missions of Configuration 1. If the electrical provisions for the rack are connected to the main system through circuitbreakers, failure of components in the rack will not cause failure of CSM equipment.

Configuration C

The reliability analysis for Apollo X is applicable to Configuration C with one major exception -- the mission success logic. During the Apollo X study, the mission was deemed to be successful if the spacecraft and crew stayed in orbit for the full 45 days. Since one fuel cell has the capability to sustain the spacecraft after the other fuel cells have failed, the reliability logic was based on the one fuel-cell requirement for mission success.

During the AES study, the experiments and power loads became more completely defined, with the result that it seems improbable that the mission could be considered successful if power loads had to be reduced to the output capability of one fuel cell, since most of the planned experiments would have to be cancelled. For this reason, the reliability logic was changed to require two fuel cells for mission success. The result of this change, shown in Figure 49, is that four 1000-hour fuel cells are required to meet the reliability goals for mission success.

Configuration D

The number of fuel cells required in the external device for a 45-day mission was calculated for both the one and the two fuel-cell mission-success requirements. The results (Figure 50) indicate that this logic difference requires the addition of at least one fuel cell. A comparison of 50 percent confidence level versus 90 percent was also made (Figure 51). As can be seen from the figures, the use of three 100-hour fuel cells in the external device rack does not quite meet the full reliability criteria, but the use of four fuel cells greatly exceeds the requirement. For this reason, the spacecraft design is based on the use of the three cells, but room is left in the external device for a fourth one if later analysis prove that it is required. The recognition that the present D external device design is marginal in this respect, emphasizes the necessity to operate SM fuel cells for the full 14 days and longer if possible. By using one fuel cell in the SM for more than 14 days and operating it in parallel with one in the external device, reliability is somewhat improved. As indicated elsewhere in this report, such operation is entirely feasible, though it is difficult to force load sharing in any way other than their natural tendency.

[redacted]

Configuration D'

The fuel cells used on Configuration D' are expected to contain some of the improvements of the 400-hour fuel cell, but may not yet be fully qualified for 1000 hours. Therefore, reliability data was generated for both the 400-hour and the 1000-hour fuel cell. In addition, the effect of mission success criteria and confidence level goals was analyzed. The results are shown in Figures 52 and 53, and indicate that more than two cells are needed in the External Device regardless of the fuel-cell type. As was the case with Configuration D, three fuel cells do not fully meet the reliability criteria, but are within the general range of error in the reliability analysis. In order to keep the external device design and instrumentation as standard as possible, the present design shows three fuel cells for this configuration. It is emphasized, however, that this analysis is based on using external device cells only after 14 days. As is indicated in the cryogenic storage system discussion, tank boiloff conditions may dictate premature start up of rack fuel cells, in which case either three 1000-hour or four 400-hour fuel cells would be required.

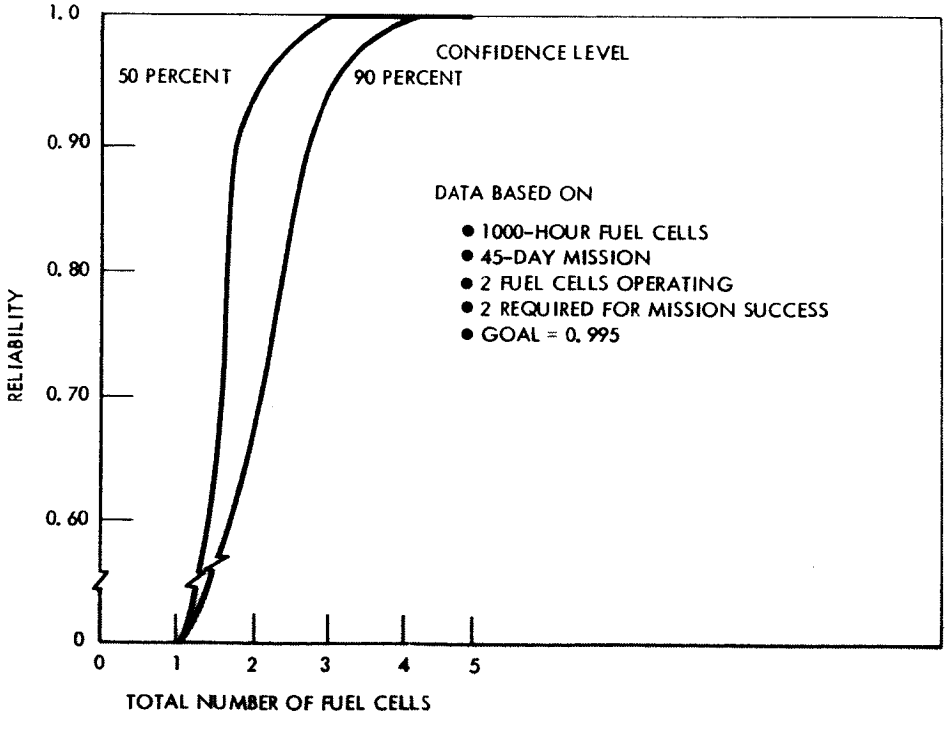


Figure 49. Configuration C Fuel-Cell Reliability

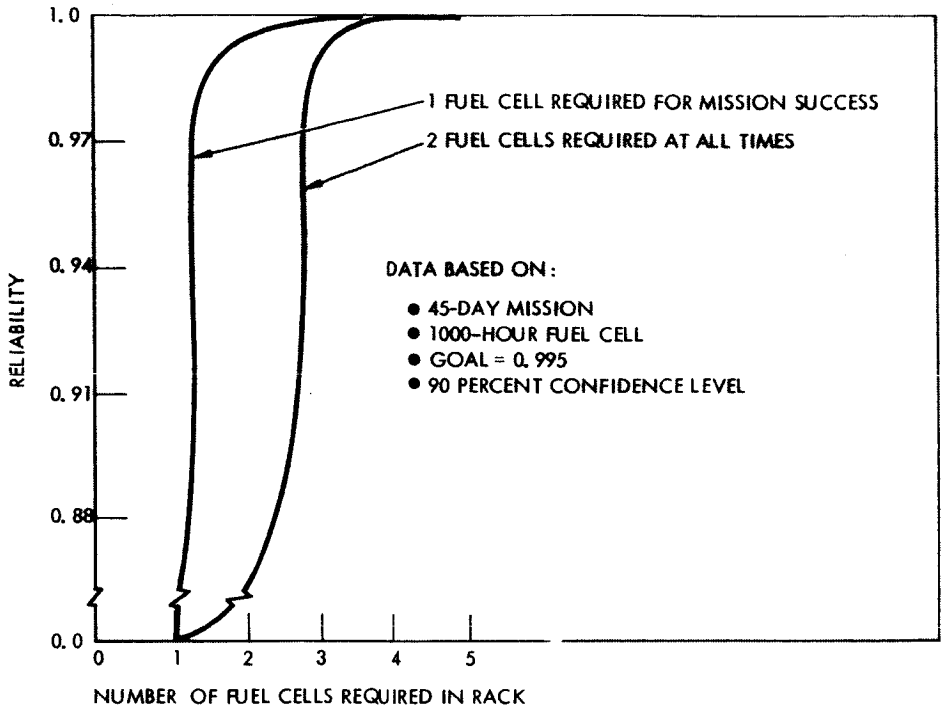


Figure 50. Configuration D Fuel-Cell Reliability (Mission Success Criterion)

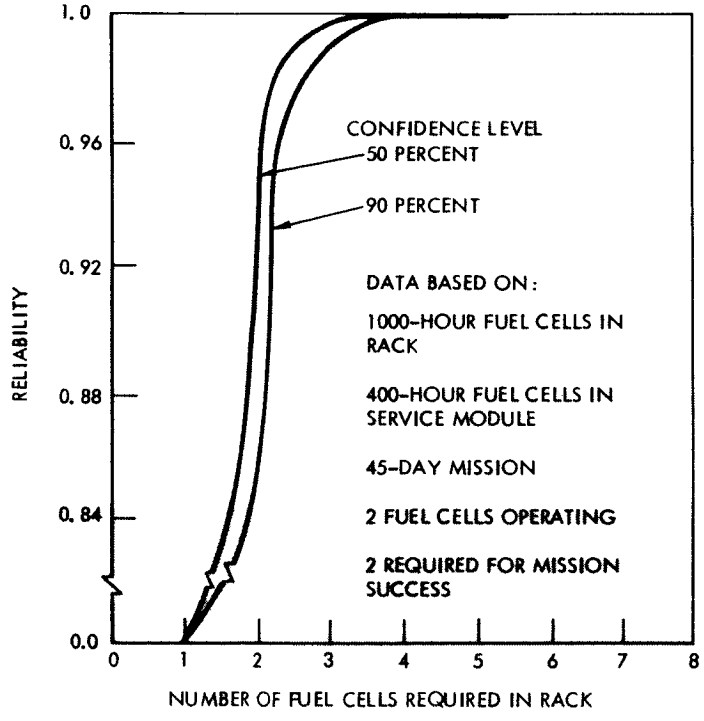


Figure 51. Configuration D Fuel-Cell Reliability (Confidence Level)

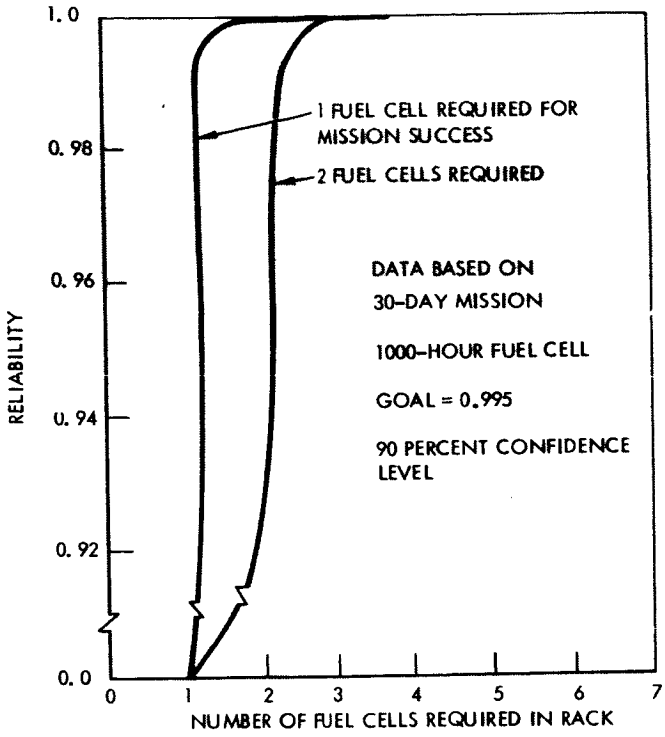


Figure 52. Configuration D Fuel-Cell Reliability (1000-Hour Cells)

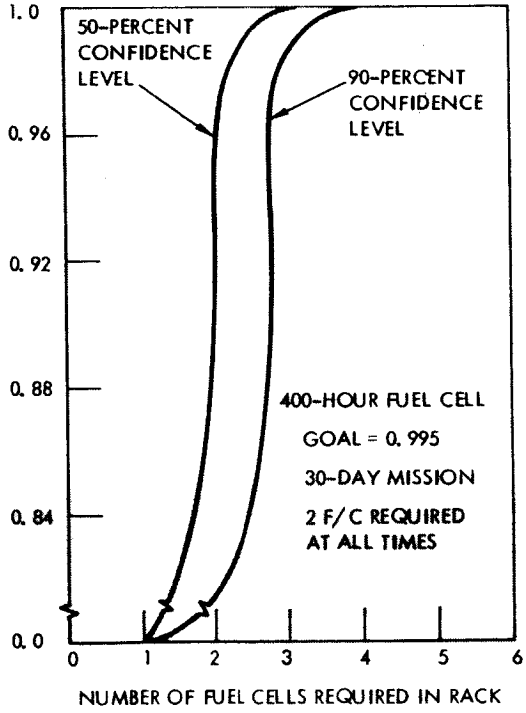


Figure 53. Configuration D' Fuel-Cell Reliability (400-Hour Cells)

INVERTERS

In each configuration a choice exists as to the source of ac power in the external device. Where it contains fuel cells, ac power is continuously required in both the external device and the CM in order to accomplish the mission. If inverters are located in the external device as well as the CM, the reliability of this system will necessarily degrade the total mission success reliability, since it is in reliability series with other essential functions. As a result, the external device inverter reliability must be very high in order not to degrade significantly the total system reliability.

The CM inverter reliability (based on one active, two standby, and mission-abort after 2 failures) is approximately 0.99452. Three inverters in the external device (again with one active, two standby, but mission-abort only after three failures) is approximately 0.99980. Thus, the installation of even three inverters in the external device degrades the reliability of having ac power (where needed) to 0.99432. No number of inverters in the external device would improve total reliability unless the inverter system reliability exceeded the reliability of redundant ac buses through the umbilical, or unless total ac loads exceeded inverter capability.

SYSTEM DESIGN

Although separate designs were developed for each of the spacecraft configurations, an attempt was made to keep sequential designs similar so that there would be a natural progression from Configuration 1 to Configuration C or from Configuration 1 to D' to D. As a result, the CM connections for umbilicals are the same for all four configurations. The number of fuel cells in the D and D' external device are the same; therefore, the fuel cell instrumentation is the same, and the instrumentation for the D' cryogenic storage system may be converted to Configuration D by removing some of the instruments.

In all 30- and 45-day configurations, fuel-cell redundancy is provided by cold fuel cells with in-space start. When the Block II CSM is providing power, three fuel cells normally operate in parallel; when power is provided by fuel cells in the external device or new fuel cells (1000 hour) in the SM, only two fuel cells normally operate at any one time. Supplementary batteries for peak loads are planned for all 30- and 45-day flights. In each configuration, all ac power is supplied from the inverters in the CM.

SPACECRAFT CONFIGURATIONS

Configuration 1 consists of the Apollo Block II CSM with tne addition of umbilicals extending both dc buses A and B and ac buses 1 and 2 into the external device. Additional silver zinc primary batteries, similar to those recommended in the Pallet study (SID 65-226), are installed in the external device as needed to increase the available energy to the amount required to perform the mission.

Configuration C replaces the three SM fuel cells with four 1000-hour cells and replaces the Block II cryogenic tanks with larger ones. A supplementary peak-power battery and associated controls are also added to the spacecraft, either in the external device or the SM. Two of the four fuel cells have in-space start provisions and batteries for the startup energy source.

Configuration D uses the Apollo Block II CSM with an additional fuel-cell power system in the external device, consisting of three 1000-hour fuel cells, each with in-space start provisions. The cryogenics are stored in new, optimized tanks. Instrumentation for the external device power system is contained and displayed in the external device. A peak-power battery system is also provided.

The one flight (Configuration D') that is scheduled before the availability of new cryogenic tanks and 1000-hour fuel cells uses Apollo Block II components entirely, with the exception of an in-space start capability added to the fuel cells. This configuration has three 400-hour fuel cells, four hydrogen tanks, and three oxygen tanks. Other provisions are the same as for Configuration D.

POWER DISTRIBUTION AND CONTROL

The power distribution and control system for each configuration is designed around the Apollo Block II configuration. The distribution and control in the CM is basically the same for all configuration whereas, for the SM, the Apollo X configuration is used in Configuration C. The external device distribution system is, of course, a new design for each concept. Presented below is a basic description of the power distribution and control system for each configuration.

Configuration 1—Command and Service Module

The electrical power distribution and control system for Configuration 1 is shown in Figure 54. Essentially, the service module and command module is the same as the present Apollo Block II, with the addition of a connection for the ac umbilical.

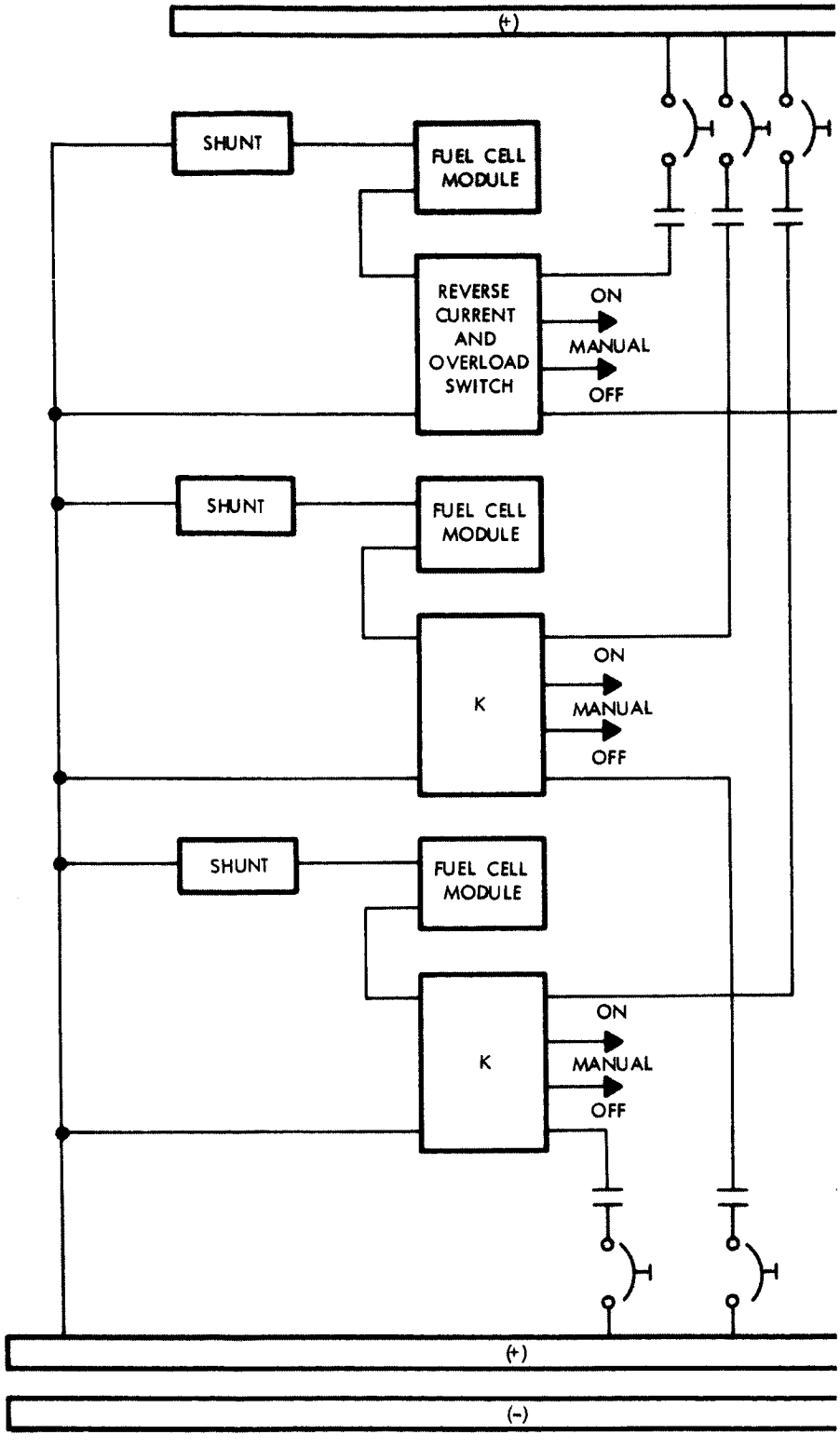
The distribution of nominal 28-vdc power in the CM is accomplished by means of a two-wire grounded system. Nominal 115 to 200 volt, 400-cps 3-phase power distribution is accomplished by means of a four-wire grounded system. Both dc and ac power will be grounded to the spacecraft structure by means of a single-point ground. The bus systems that comprise the power distribution system are as follows:

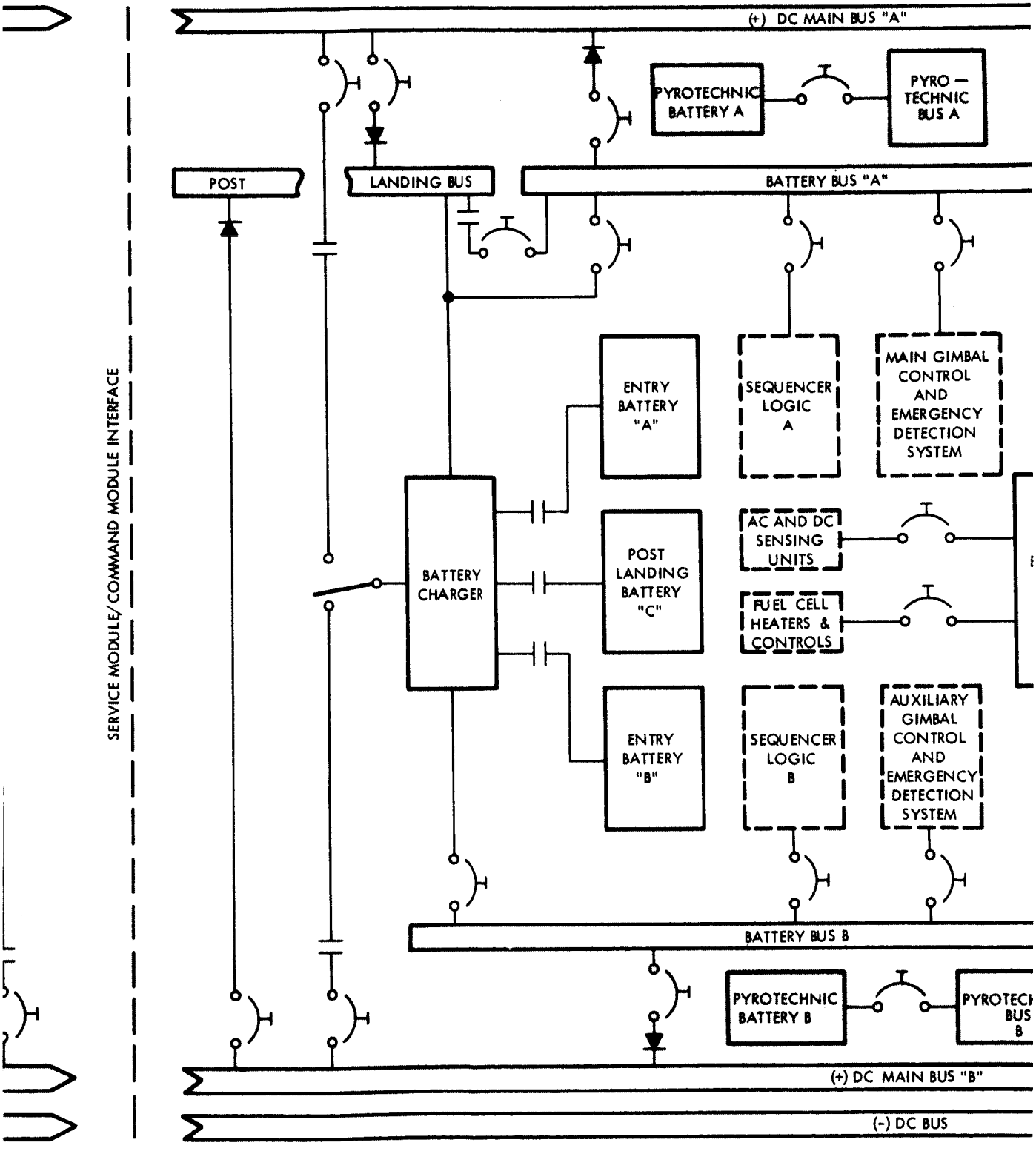
Main Buses A and B—The main dc bus system is comprised of main dc bus A and main dc bus B—two isolated and redundant positive buses that receive their power from the three fuel-cell modules. Each fuel-cell output is connected in parallel to both buses. Connection between the fuel cells and SM buses A and B is accomplished by utilizing motorized power transfer switches.

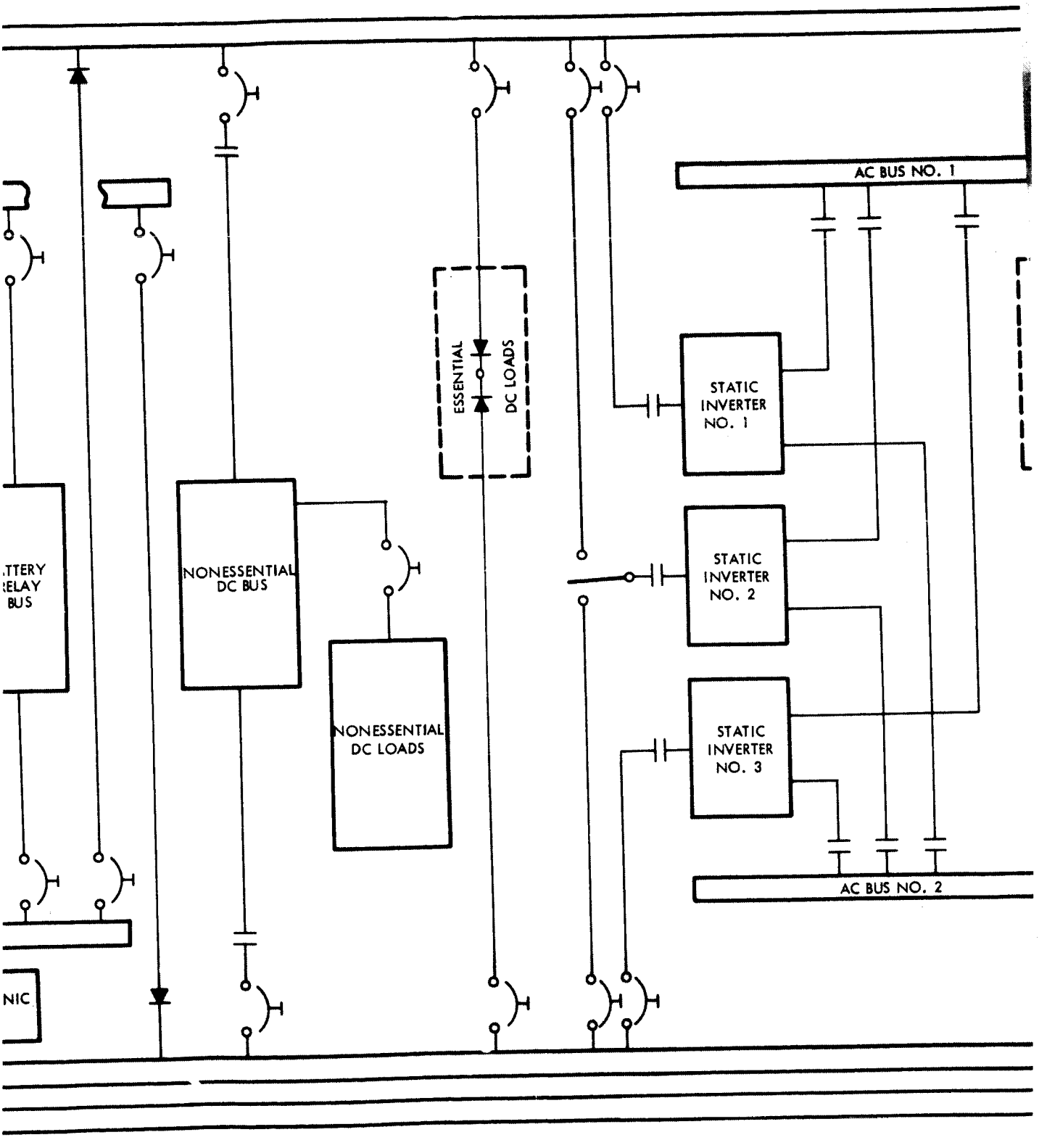
The buses supply nominal 28-vdc power to the electric load system. The loads are identified and connected to the dc buses as follows:

Nonredundant essential loads—each nonredundant essential load is connected to both dc buses. The buses are isolated from each other by blocking diodes.

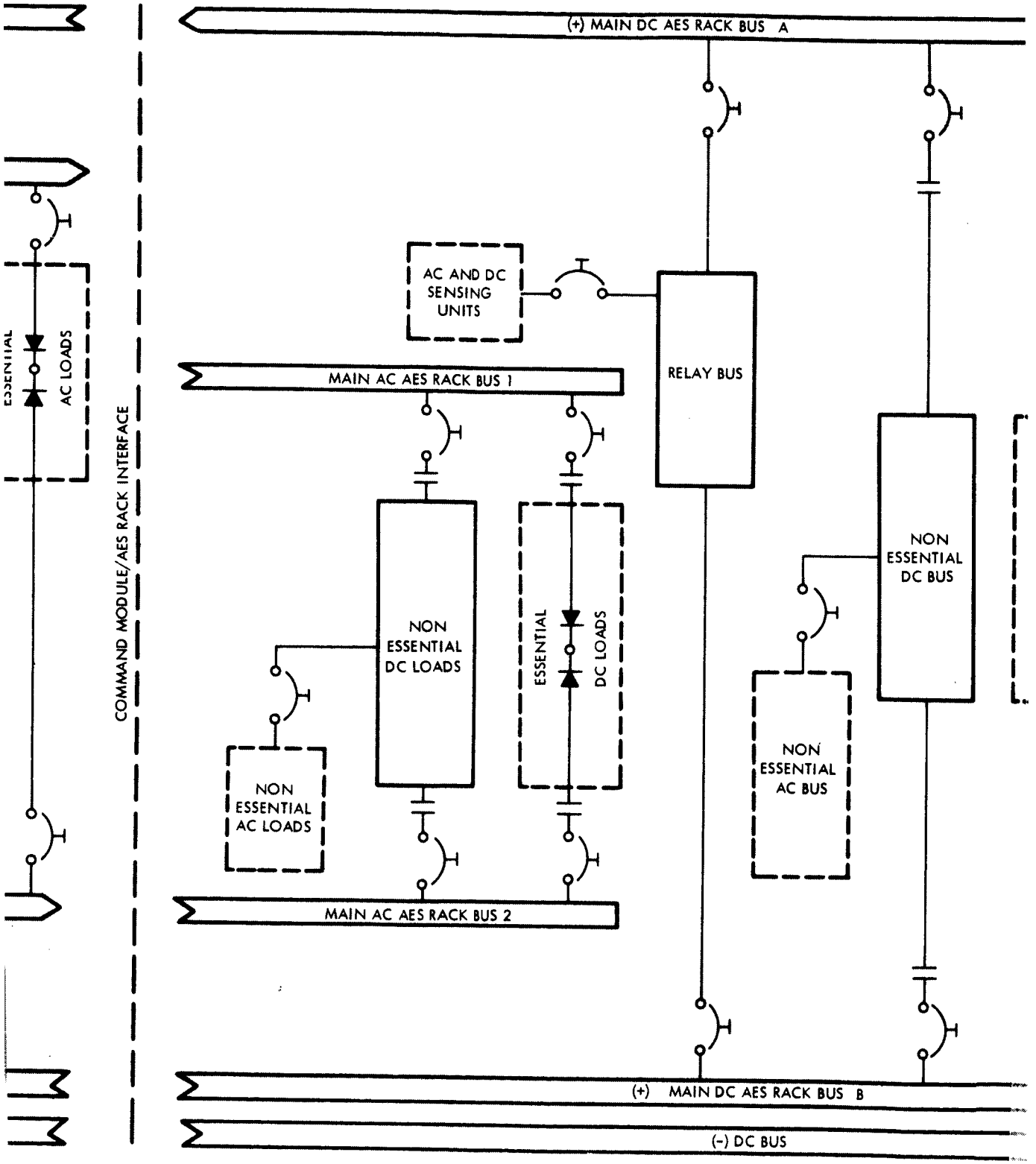
Redundant Essential Loads—Redundant essential loads are connected alternately to main bus A or B.


(+)
SHUNT
FUEL CELL MODULE
REVERSE CURRENT AND OVERLOAD SWITCH
ON
MANUAL
OFF
SHUNT
FUEL CELL MODULE
K
ON
MANUAL
OFF
SHUNT
FUEL CELL MODULE
K
ON
MANUAL
OFF
(+)
(-)


(+) DC MAIN BUS "A"
PYROTECHNIC BATTERY A
PYRO – TECHNIC BUS A
POST
LANDING BUS
BATTERY BUS "A"
ENTRY BATTERY "A"
SEQUENCER LOGIC A
MAIN GIMBAL CONTROL AND EMERGENCY DETECTION SYSTEM
BATTERY CHARGER
POST LANDING BATTERY "C"
AC AND DC SENSING UNITS
FUEL CELL HEATERS & CONTROLS
ENTRY BATTERY "B"
SEQUENCER LOGIC B
AUXILIARY GIMBAL CONTROL AND EMERGENCY DETECTION SYSTEM
BATTERY BUS B
PYROTECHNIC BATTERY B
PYROTECH BUS B
(+) DC MAIN BUS "B"
(-) DC BUS
SERVICE MODULE/COMMAND MODULE INTERFACE


AC BUS NO. 1
STATIC INVERTER NO. 1
STATIC INVERTER NO. 2
STATIC INVERTER NO. 3
AC BUS NO. 2
ESSENTIAL DC LOADS
NONESSENTIAL DC BUS
NONESSENTIAL DC LOADS
TTERY
ELAY
BUS
NIC

CONFIDENTIAL


(+) MAIN DC AES RACK BUS A
AC AND DC SENSING UNITS
RELAY BUS
MAIN AC AES RACK BUS 1
NON ESSENTIAL DC LOADS
ESSENTIAL DC LOADS
NON ESSENTIAL DC BUS
NON ESSENTIAL AC LOADS
NON ESSENTIAL AC BUS
MAIN AC AES RACK BUS 2
(+) MAIN DC AES RACK BUS B
(-) DC BUS
AC LOADS
COMMAND MODULE/AES RACK INTERFACE

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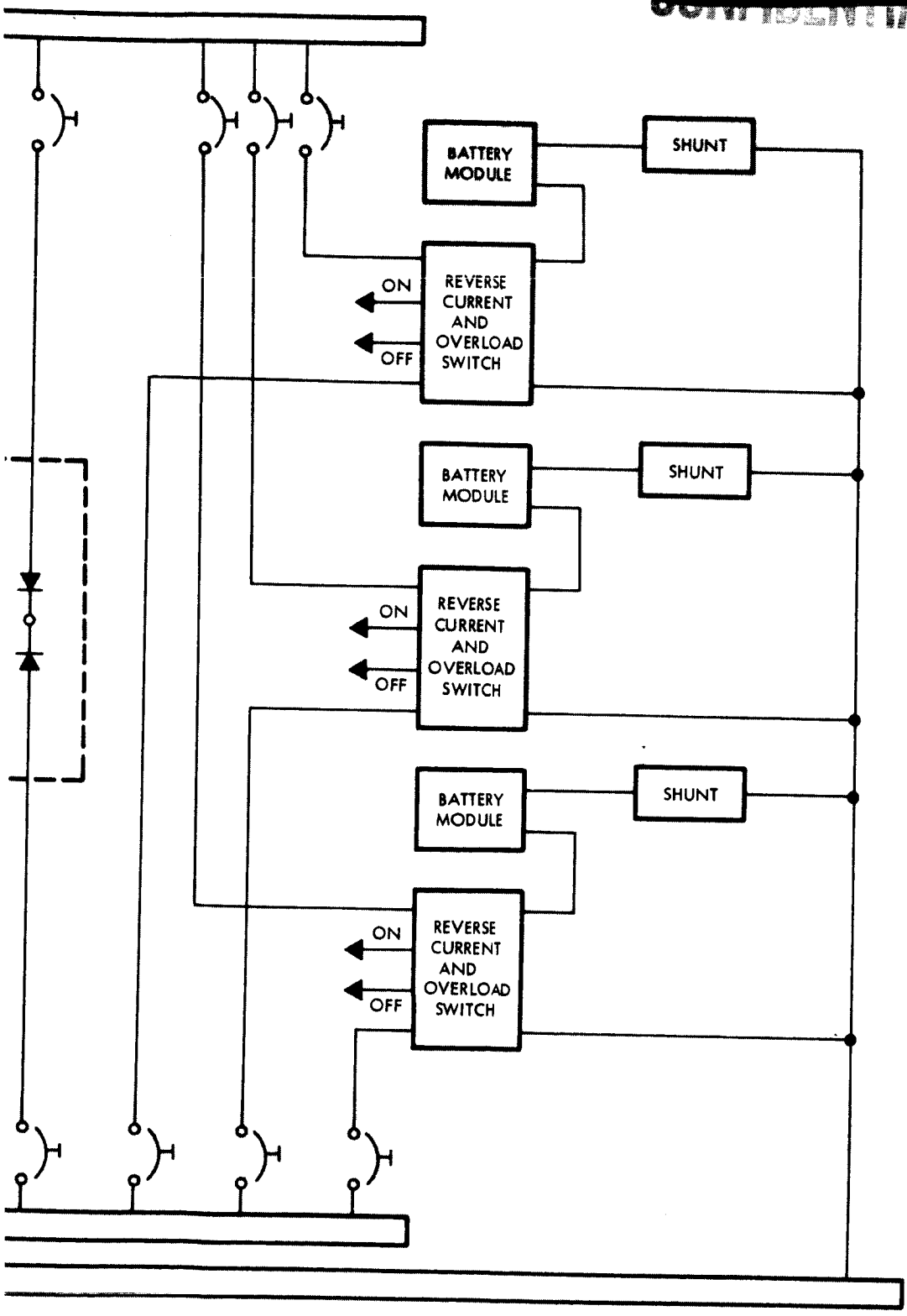

Figure 54. Configuration 1 Power System Schematic

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The buses are located in the service and command modules. Main bus A in the SM is connected to main bus A in the CM by eight 12-gauge power feeders. Main bus B in the SM is also connected to main bus B in the CM by eight 12-gauge power feeders. Each set of power feeders goes through a separate pyrotechnic power interrupter switch. The power interrupter dead end switches are located in the CM.

AC Buses 1 and 2—The ac buses 1 and 2 receive their power from the electrical power system's three static inverters. The buses supply 115/200 volt, 400-cps, 3-phase power to the SC ac-load system.

The ac buses are connected to the static inverters output by utilizing motorized power transfer switches. These motor switches are operated by dc power from the battery relay bus.

Conditions that must be met when connecting the inverters to the buses are as follows: (1) One inverter may be connected to both buses, (2) two inverters may be connected to separate buses, (3) two inverters may not be connected to one bus, (4) three inverters may not be connected to either or both buses.

Inverter System—The Apollo spacecraft static power inverter is a semiconductor device used to convert spacecraft dc power to ac power. The inverter converts 28-vdc power to 115/200 volt (rms) 400-cps 3-phase ac power, and has an output rating of 1250 volt-amperes. The function of the inverter is to supply all the ac power required by the spacecraft.

Configuration 1—External Device

Primary silver-oxide-zinc batteries connected in parallel will supplement fuel-cell power by means of redundant dc buses. Each primary battery can either be manually or automatically isolated from the main dc bus in case of battery malfunction. The output control of each battery is through an overload and reverse current-sensing device similar to that presently utilized in the fuel-cell outputs. Normally, the batteries will only be connected to the bus during periods of peak power demand and after the SM reactant supply has been depleted.

The dc bus system, relay bus, and sensing devices are basically the same as those described for the CM. The one major difference between the external device distribution system and that of the command module is the supplying of ac power through the CM inverters and its distribution through an umbilical.

Configuration C—CSM and External Device

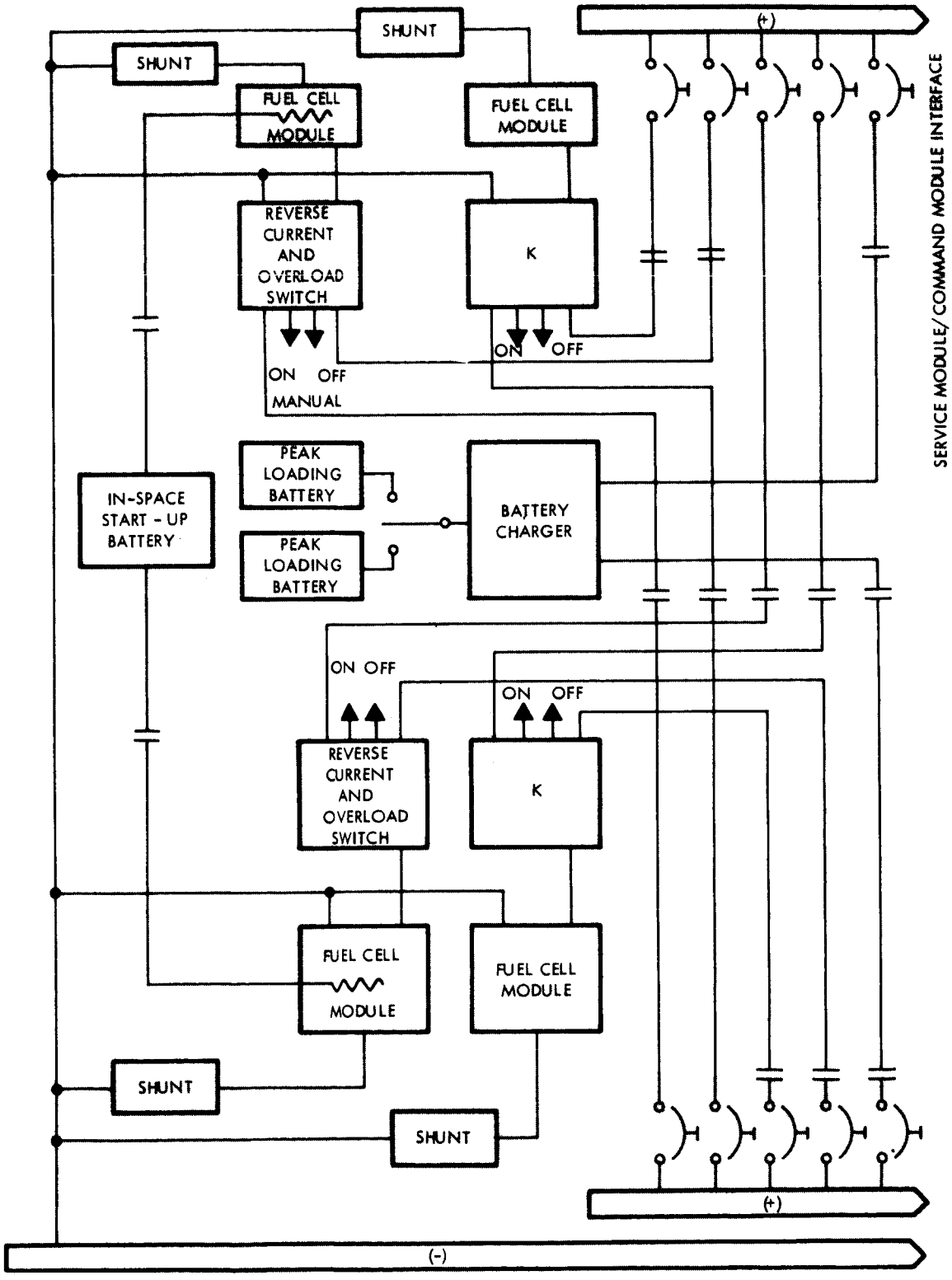
Configuration C (Figure 55) is essentially the Apollo X configuration, consisting of four 1000-hour fuel cells in the SM in place of the present three 400-hour Apollo Block II modules. Two of the four modules operate in parallel handling system loads, while the other two modules have setup startup capability to be utilized whenever required. The in-space startup battery is located in the service module; additional secondary peak loading batteries could be located in the external device or in the SM. The power distribution is essentially the same as that of configuration 1 except that the power makeup batteries in the external device have been deleted.

Configuration D—CSM and External Device

Configuration D, Shown in Figure 56, differs from that of Configuration 1 in that three 1000-hour fuel-cell modules replace the primary batteries in the external device. Each fuel cell is capable of in-space startup, and the in-space startup secondary battery is also located in the external device along with the secondary peak loading batteries.

Configuration D'—CSM and External Device

The power system is the same as for Configuration D, with minor exceptions due to the different number of cryogenic tanks and any changes in fuel-cell start and operation between 400-hour and 1000-hour fuel cells.


SHUNT
SHUNT
FUEL CELL MODULE
FUEL CELL MODULE
REVERSE CURRENT AND OVERLOAD SWITCH
K
ON OFF
ON OFF MANUAL
PEAK LOADING BATTERY
PEAK LOADING BATTERY
IN-SPACE START - UP BATTERY
BATTERY CHARGER
ON OFF
ON OFF
REVERSE CURRENT AND OVERLOAD SWITCH
K
FUEL CELL MODULE
FUEL CELL MODULE
SHUNT
SHUNT
(+)
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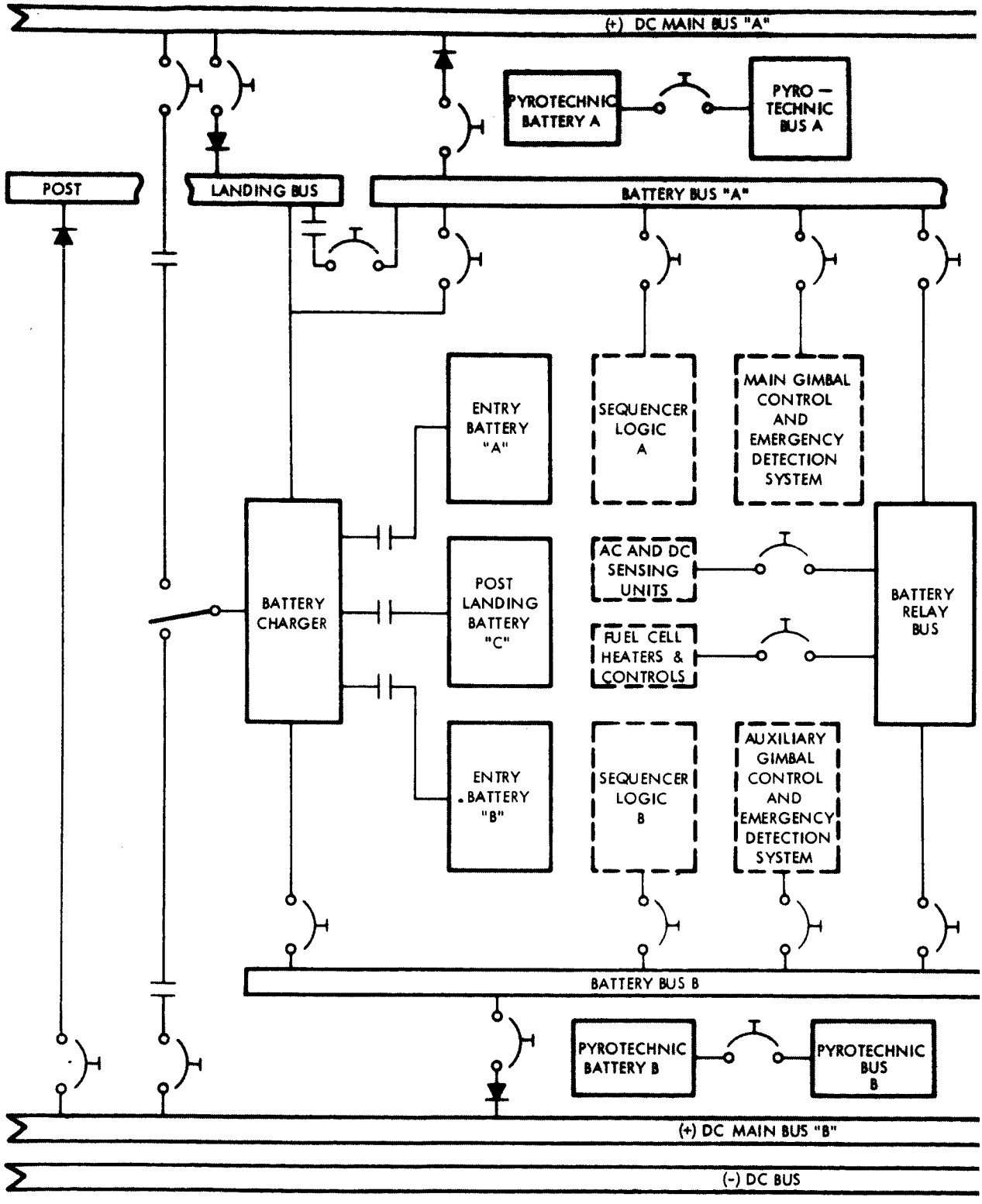
(-)
SERVICE MODULE/COMMAND MODULE INTERFACE


(+) DC MAIN BUS "A"
PYROTECHNIC BATTERY A
PYRO – TECHNIC BUS A
POST
LANDING BUS
BATTERY BUS "A"
ENTRY BATTERY "A"
SEQUENCER LOGIC A
MAIN GIMBAL CONTROL AND EMERGENCY DETECTION SYSTEM
BATTERY CHARGER
POST LANDING BATTERY "C"
AC AND DC SENSING UNITS
FUEL CELL HEATERS & CONTROLS
BATTERY RELAY BUS
ENTRY BATTERY "B"
SEQUENCER LOGIC B
AUXILIARY GIMBAL CONTROL AND EMERGENCY DETECTION SYSTEM
BATTERY BUS B
PYROTECHNIC BATTERY B
PYROTECHNIC BUS B
(+) DC MAIN BUS "B"
(-) DC BUS

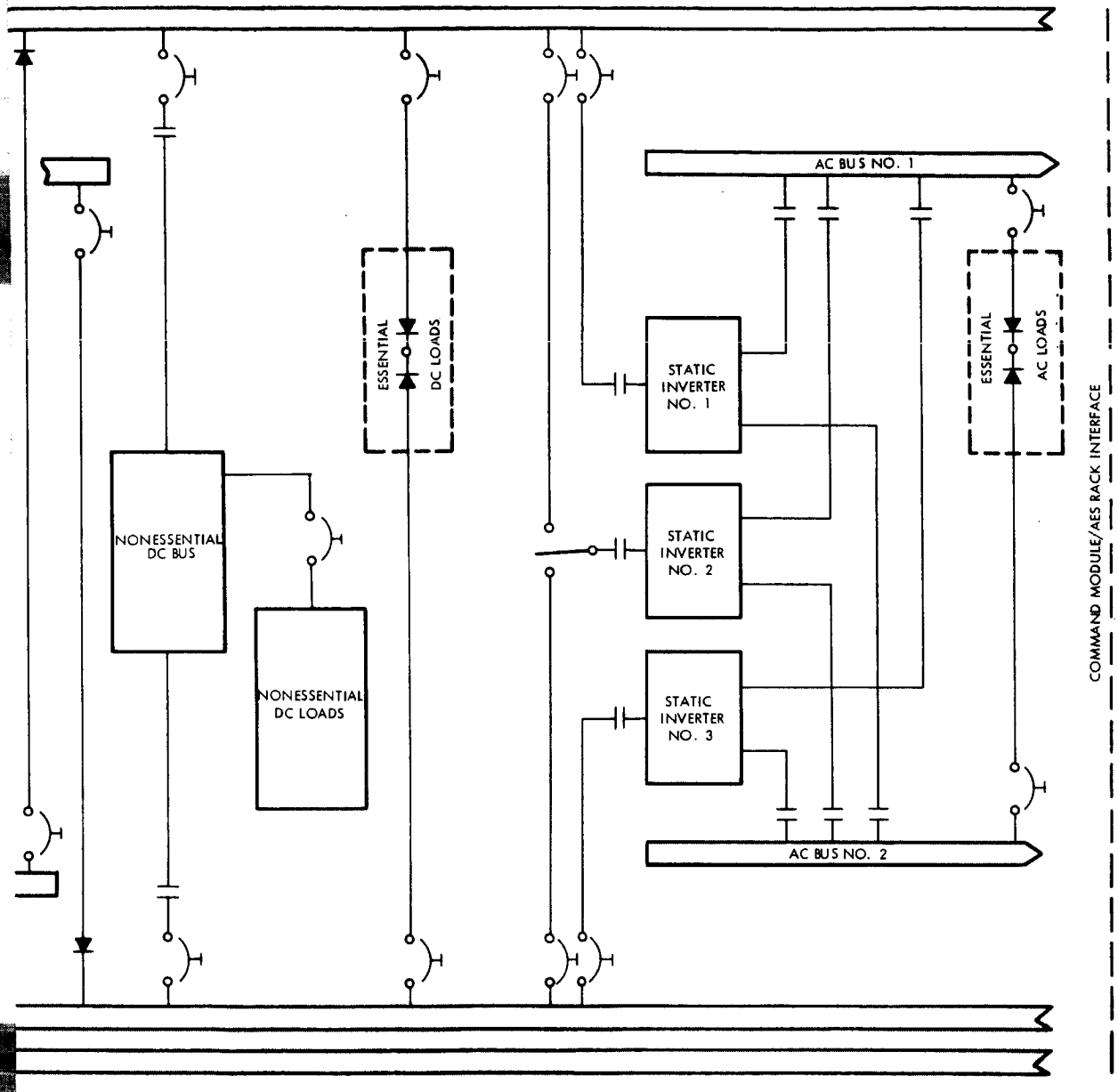
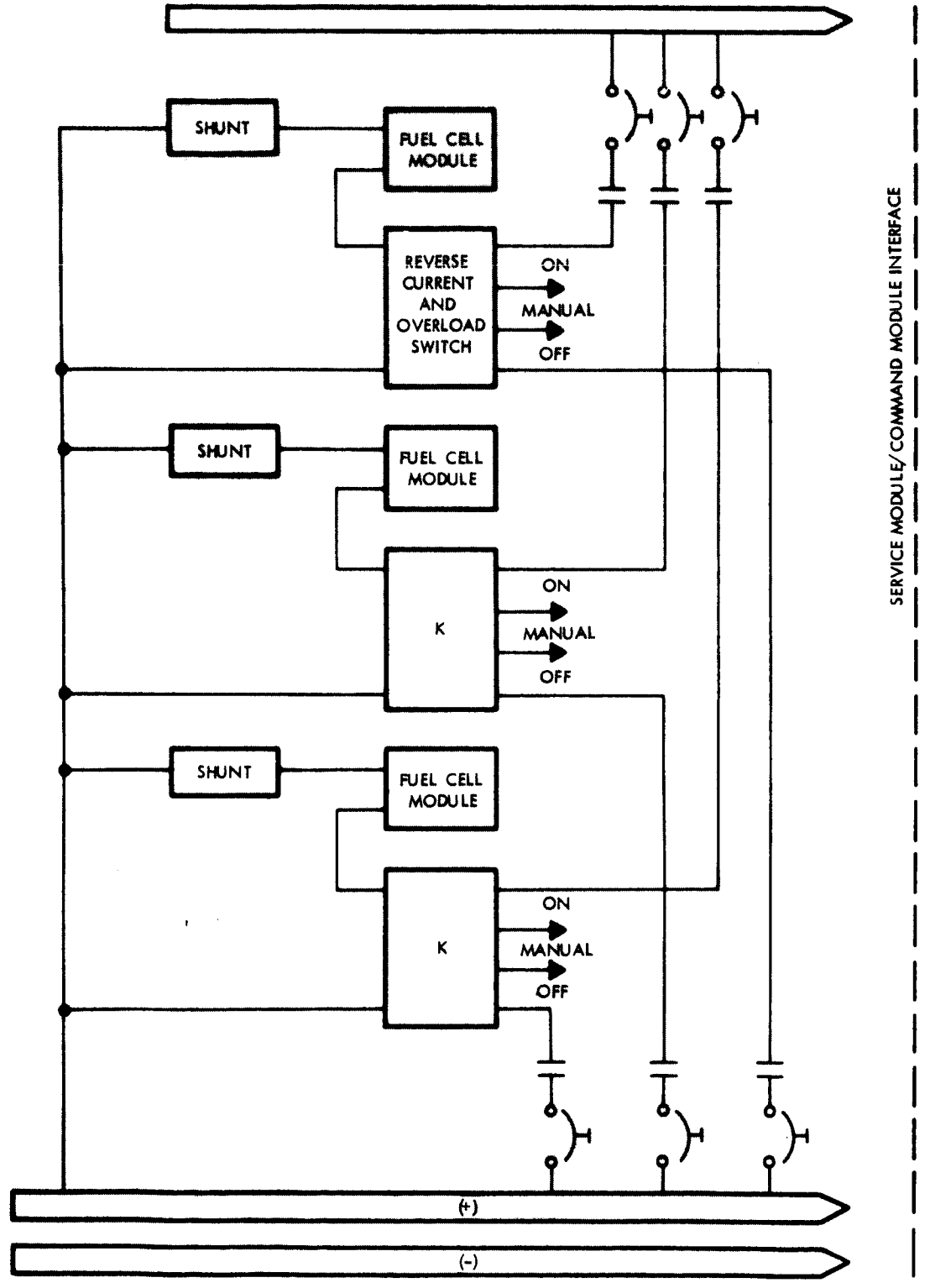


Figure 55. Configuration C Power System Schematic


SHUNT
FUEL CELL MODULE
REVERSE CURRENT AND OVERLOAD SWITCH
ON
MANUAL
OFF
SHUNT
FUEL CELL MODULE
K
ON
MANUAL
OFF
SHUNT
FUEL CELL MODULE
K
ON
MANUAL
OFF
(+)
(-)
SERVICE MODULE/COMMAND MODULE INTERFACE

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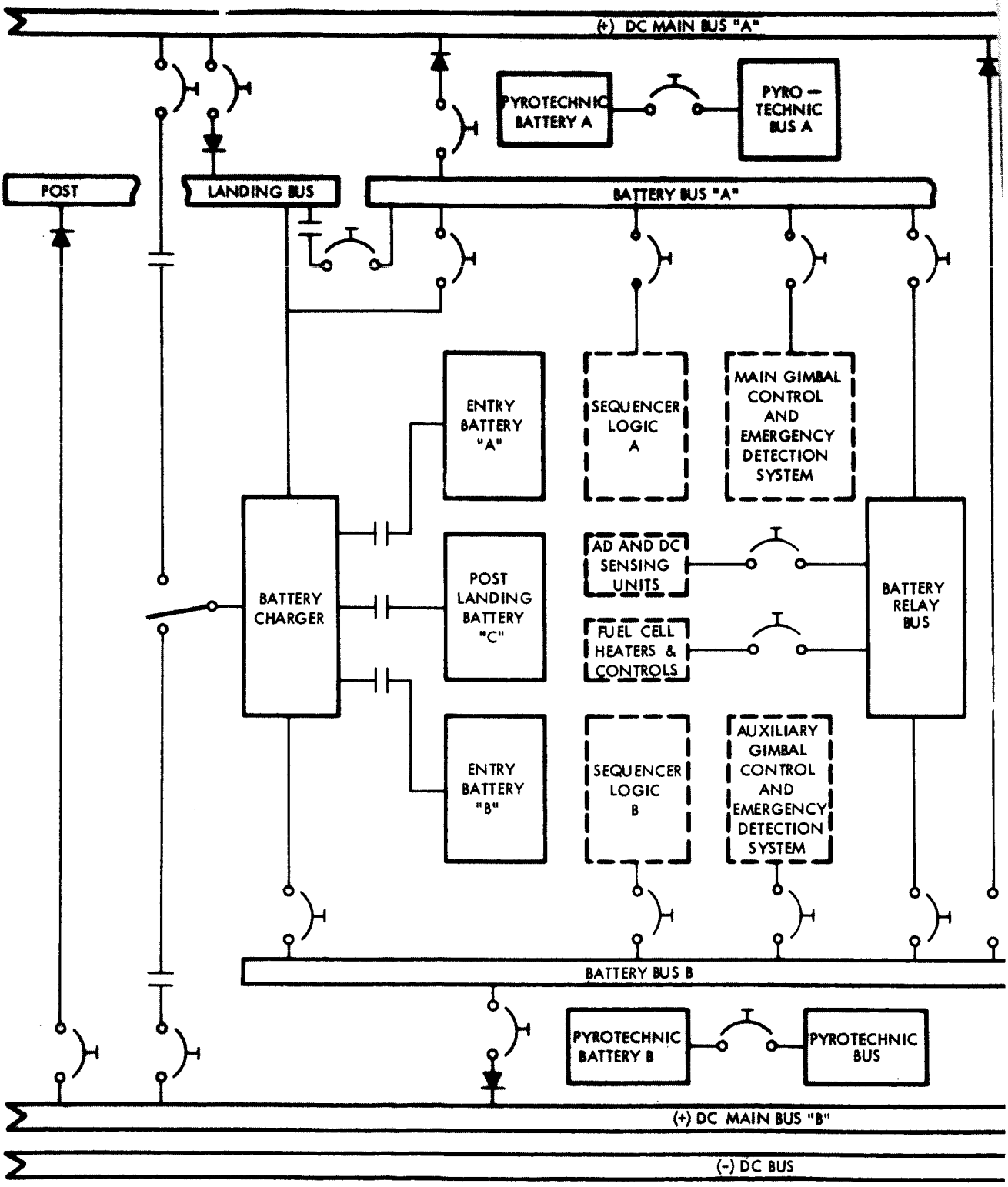

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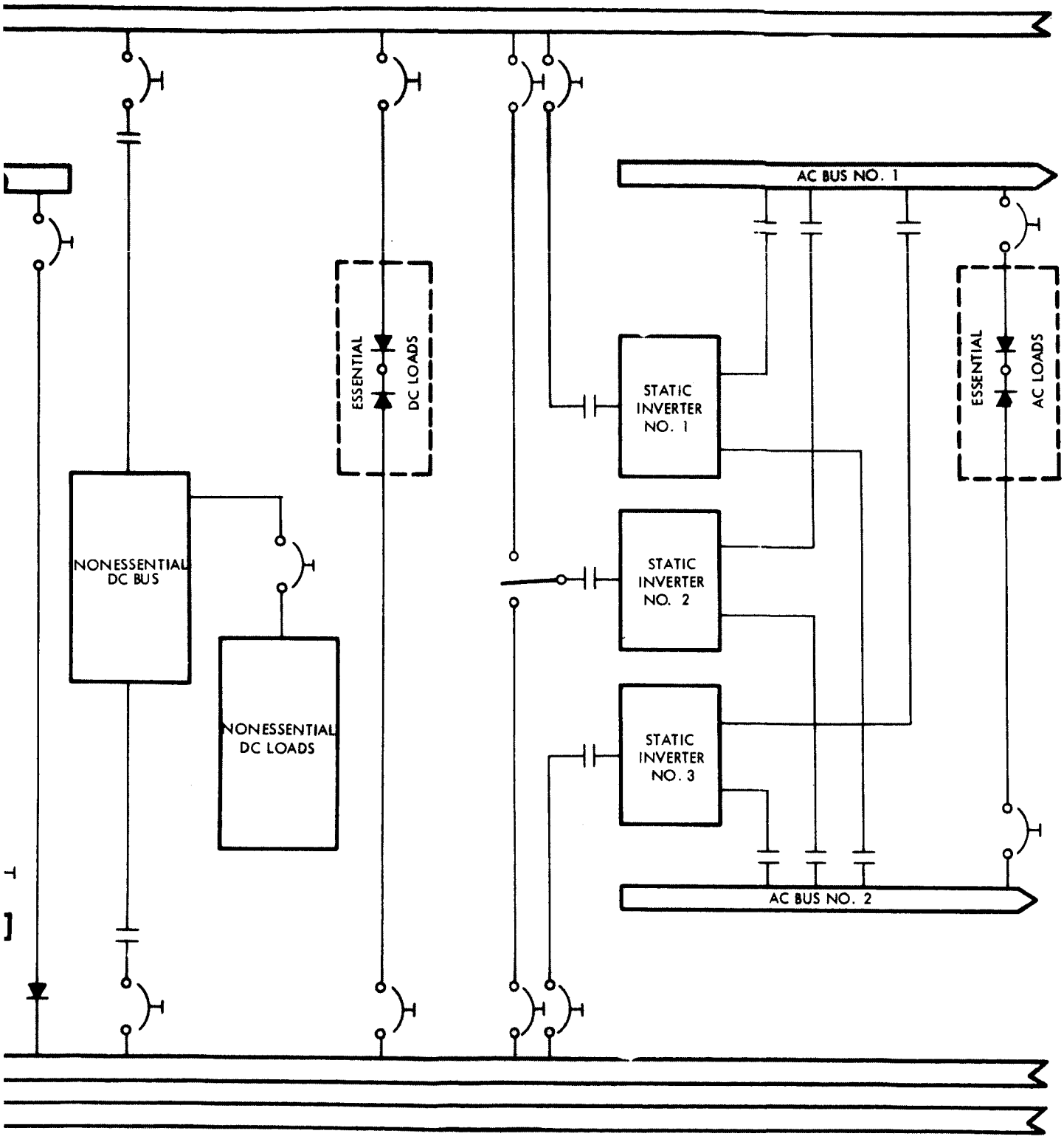

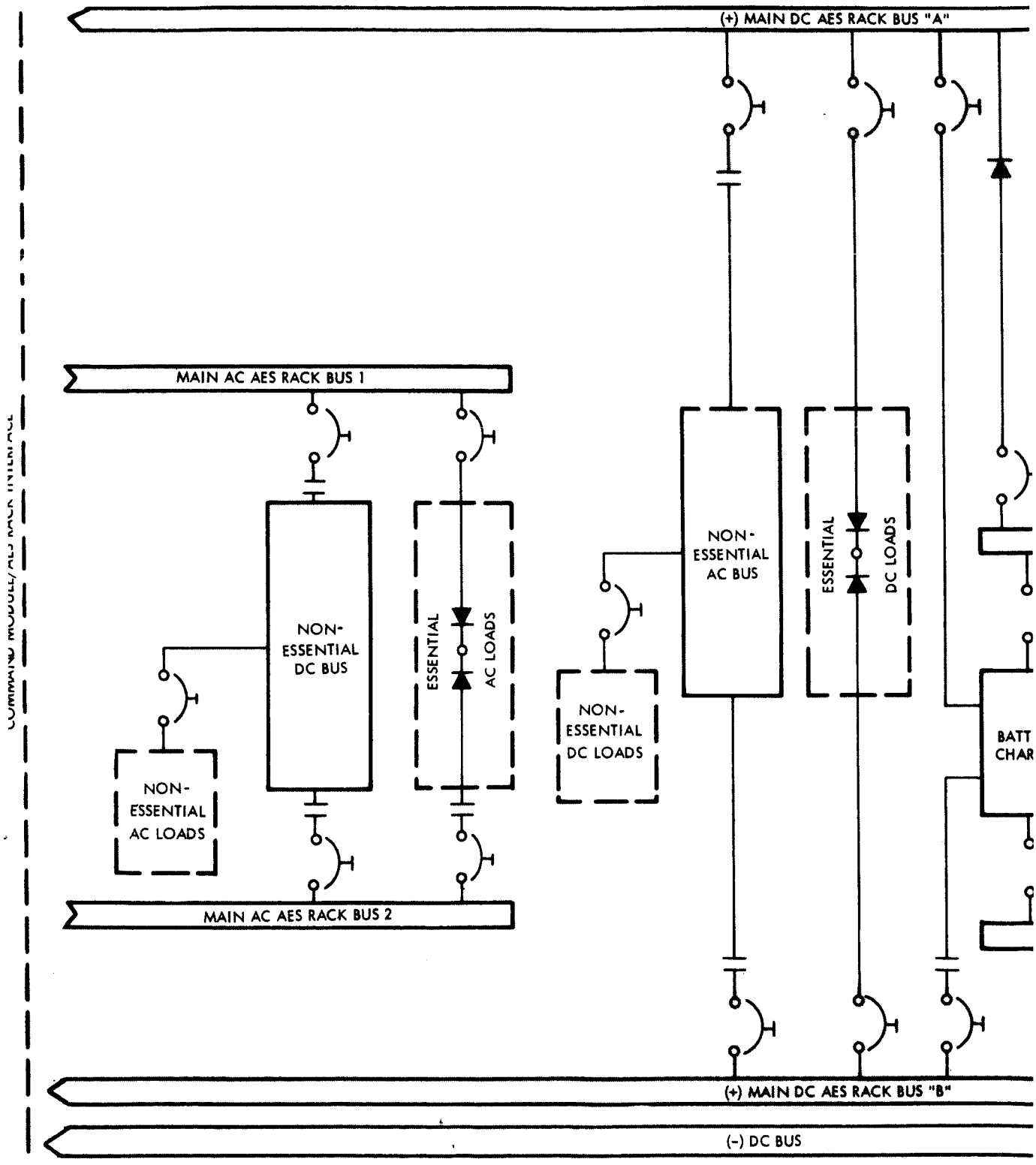
(+) MAIN DC AES RACK BUS "A"
MAIN AC AES RACK BUS 1
NON-
ESSENTIAL
DC BUS
ESSENTIAL
AC LOADS
NON-
ESSENTIAL
AC LOADS
MAIN AC AES RACK BUS 2
NON-
ESSENTIAL
DC LOADS
NON-
ESSENTIAL
AC BUS
ESSENTIAL
DC LOADS
BATT
CHAR
(+) MAIN DC AES RACK BUS "B"
(-) DC BUS

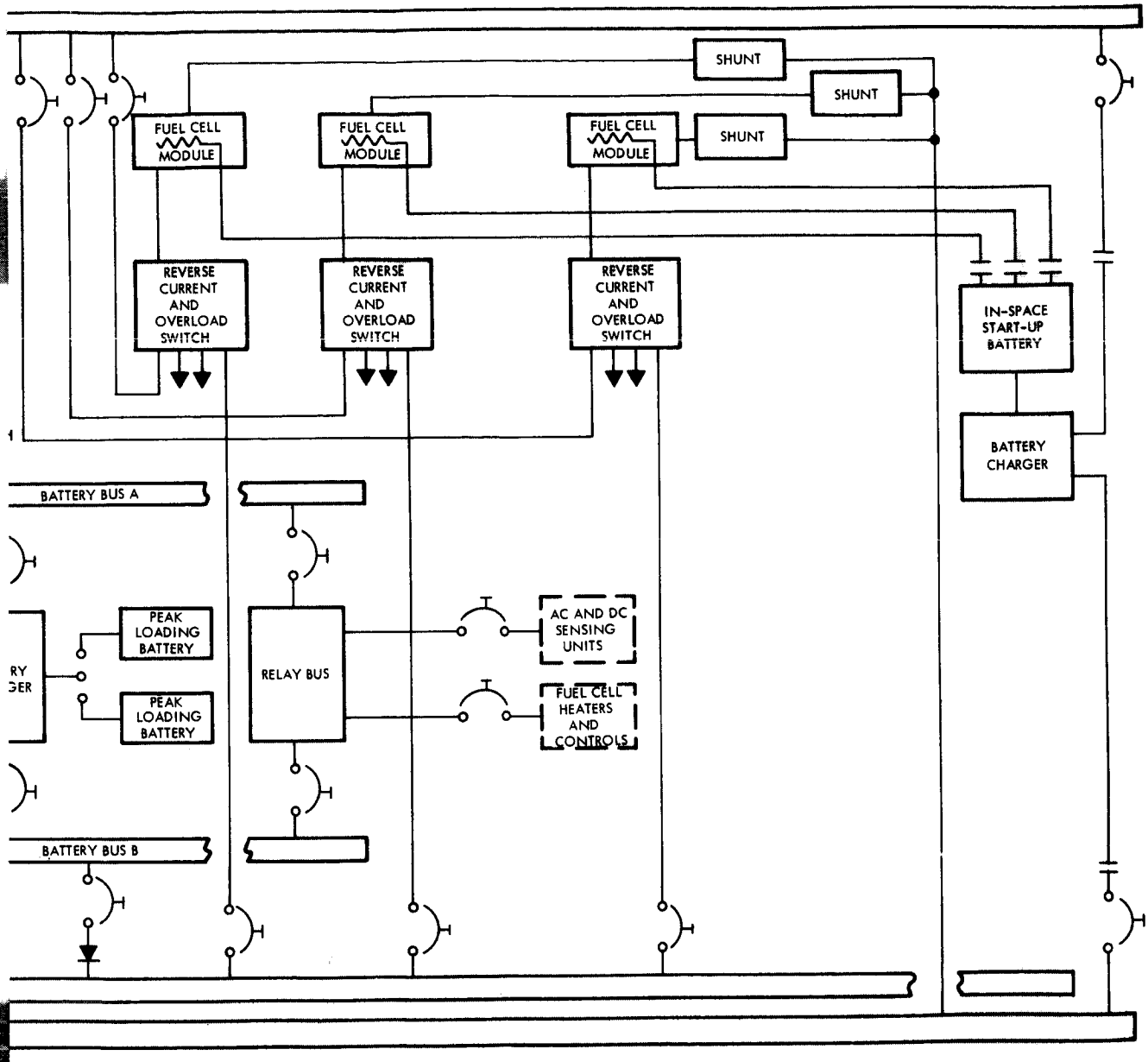

Figure 56. Configurations D and D' Power System Schematic

CRYOGENIC STORAGE SYSTEMS

Early in the study it became evident that the cryogenic storage subsystem was critical in terms of mission performance and duration. This criticality was evident in terms of reduced power generation capability, airlock repressurization capability, metabolic oxygen, and payload weight. These items, in turn, translate into reduced mission duration and experimental performance capability.

The cryogenic storage subsystem is required to store and release, as needed, hydrogen and oxygen to the power system fuel cells and metabolic and pressurization oxygen to the environmental control system. The study ground rules dictated the use of the Apollo Block II system in the service module of Configuration 1, D, D', and multiple Block II cryogenic tanks in the external device for Configuration D'. It was permissible to use the Apollo X concept cryogenic storage system in the service module of Configuration C and in the external device for Configuration D.

BLOCK II CRYOGENIC STORAGE SYSTEM

This system has been described in detail in Project Apollo reports; however, a brief description of the system components, shown in Figure 57, is presented here for reference. Both hydrogen and oxygen are stored in the supercritical state. Each hydrogen subsystem and oxygen subsystem consists of:

1. Two insulated tank assemblies, each with a density and temperature probe, two fan motors, two heat elements, and a filter
2. A tank valve module with components for individual control of both tanks
3. A fuel cell valve module for fluid control to each fuel cell from each tank
4. One fill and one vent coupling per tank, and a quick-disconnect coupling for purging
5. An electrical connector for each tank
6. A signal conditioner for each tank controlling density and temperature readout.

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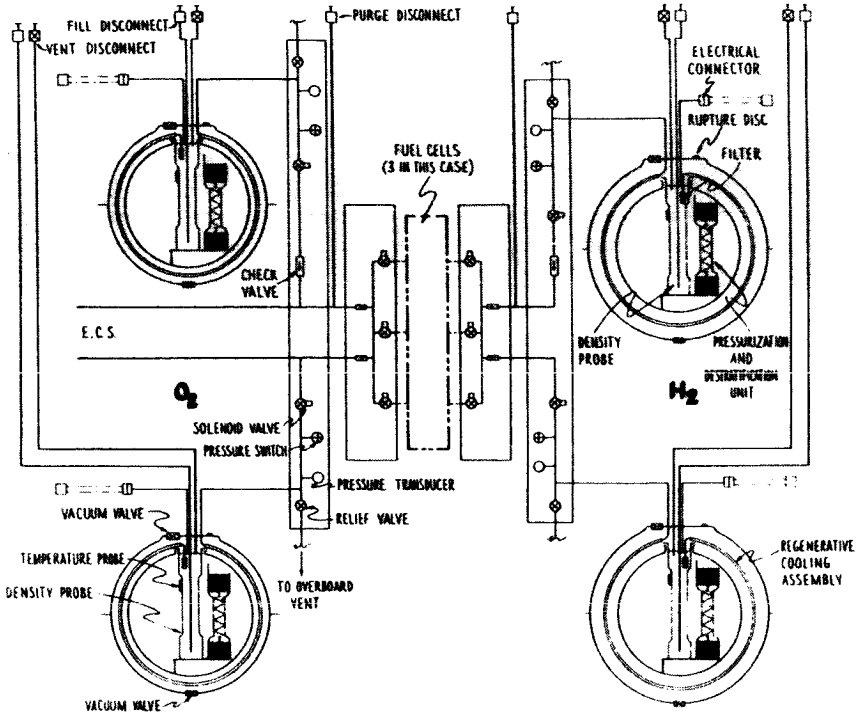

Figure 57. Apollo Systems Schematic

APOLLO X CRYOGENIC STORAGE SYSTEM

During the previous Apollo X study a new cryogenic storage system was defined based on load estimates made at that time. These estimates were deliberately conservative, resulting in an estimated average hydrogen demand of 0.19 lb/hr and an oxygen demand of 2.22 lb/hr. The resulting physical size of these tanks is close to the maximum that can reasonably be fitted into Sectors I and IV of the SM and still leave room for plumbing, access, and the fuel cells. The characteristics of these tanks and the Apollo Block II tanks were used exclusively during the AES study (rather than attempting to optimize a new system), since the Apollo X tanks are adequate for each of the missions as presently defined. If loads increase, the tank capacity can be increased slightly by increasing the diameter about three inches and by lowering the temperature (thereby increasing the density) of the fluid at the time of loading. This latter technique is presently being considered by Apollo and consists of loading the tanks at a higher pressure than presently planned, thereby achieving operating pressure partly by GSE pressure rather than almost completely by expansion of the fluid. (Cryogenic quantities and therefore available energy from the Block II SM is based on the use of this loading technique.)

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A comparison of the characteristics of cryogenic storage system tank designs used in this study is made in Table 36. These characteristics are expected to be nearly the same whether the tanks are located in the SM or in the external device.

Table 36. Block II and Apollo X Cryogenic Tanks

Characteristic	Block II		Apollo X	
	Hydrogen	Oxygen	Hydrogen	Oxygen
Operating pressure (psia)	250	900	250	900
Burst pressure (psia)	450	1530	450	1530
Usable fluid (lb/tank)	28	320	102.5	1198.5
Total fluid (lb/tank)	29.12	326.4	104.5	1210.5
Number of tanks	2	2	2	2
Heat leak (Btu/hr)	8.0*	19.3*	9.5**	31.0**
Dry weight (lb)	178.4	179.4	343	505
Wet weight (lb)	236.64	832.2	552	2926
Pressure vessel size (in.)	28.3 Sphere	25.1 Sphere	36.3 dia 53.6 long	36.4 dia 41.7 long
Outer shell size (in.)	31.8 Sphere	26.5 Sphere	40.3 dia 57.7 long	40.3 dia 45.6 long

(*) $_{30\,F}$ (**) $_{150\,F}$

REQUIREMENTS

The energy requirements for the various missions are made up of the vehicle loads and the experiment loads. The total energy requirements for each mission are shown in the power system section of this report.

CONFIGURATION 1 (14 DAYS)

This configuration makes use of the Block II cryogenic storage syste. The total energy capacity from the Block II fuel cells is 620 kwh. Any additional requirements are supplied by batteries in the external device. This capability utilizes the entire 56 pounds of H_2 and 448 pounds of O_2. Metabolic O_2 adds 84 pounds and leakage 98 pounds to make a total O_2 requirement of 630 pounds. This makes 10 pounds of O_2 available for air lock repressurization (two at 5 pounds per repressurization). All additional repressurizations will come from a high pressure O_2 storage bottle in the external device.

CONFIGURATION D' (30 DAYS)

The requirement defined was for 1358 kwh of electrical energy and its O_2 and H_2 expenditure. To this are added the requirements for 180 pounds of Metabolic O_2 and 210 pounds of O_2 to replace leakage.

CONFIGURATION D (45 DAYS)

Electrical energy requirements varied from 1863 kwh on Air Force Flight 2 to 2521 kwh for Flight 219. Metabolic and leakage O_2 requirements were 270 and 315 pounds respectively.

CONFIGURATION C (45 DAYS)

Electrical energy requirements varied from 1820 kwh for Air Force Flight 2 to 2478 kwh for Flight 219. Metabolic and leakage O_2 requirements are the same as for Configuration C.

CRYOGENIC STORAGE SYSTEM DEFINITION

Of the four configuration concepts, Configuration D and D' present the most critical problem areas. This section discusses Configuration D' first, although boiloff problems encountered for D' also apply to D. However, the solution for Configuration D is simpler and easier to implement. For Configurations 1 and C the cryogenics are located in the SM and are in use from pre-launch throughout the mission. These configurations present no boiloff problems.

CONFIGURATION D'

Configuration D', with its 30-day mission duration, presents a unique situation since the larger Apollo X tanks cannot be available in time for the projected flight date. To eliminate this scheduling problem multiples of the Block II tanks are installed in the external device. During the first 14 days of the mission the SM tanks are used, with the external device tanks in a standby condition. However, with cryogenic fluids, the heat-leak from the environment into the tank causes an increase of pressure until the tank must be vented or rupture will occur. A study was therefore made of the pressure buildup and venting characteristics of the Block II tanks to determine the no-loss standby time and the vent rates from heat-leak alone.

For a tank of volume V, cubic feet with an initial fill of W pounds, the total heat, Q, in Btu which can be absorbed before venting is

$$Q = W (h_2 - h_1) - \frac{144\ V (P_2 - P_1)}{J}$$

The values of h_2 and h_1 were taken from a pressure-enthalpy chart at the venting pressure P_2 and initial pressure P_1, respectively. The initial fill is usually a mixture of liquid and vapor; thus, h_1 must be based on the quality X, the latent heat L, and the saturated liquid enthalpy h_{s1}:

$$h_1 = h_{s1} (1 - X) + X h_V$$

$$h_V = h_{s1} + L$$

$$\therefore h_1 = h_{s1} (1 - X) + X (h_{s1} + L)$$

and

$$h_1 = h_{s1} + XL$$

The quantity X can be found from a material balance

$$W = W_L + W_V = W(1 - X) + WX$$

$$V = V_L + V_V = W_L v_L + W_V v_V$$

$$V = W(1 - X) v_L + WX v_V$$

$$\frac{V}{W} = (1 - X) v_L + X v_V = v_L - X v_L + X v_V$$

and

$$X = \frac{\frac{V}{W} - v_L}{v_V - v_L}$$

Values of Q for various fill weights W were calculated, and using the design heatleak of 19.3 Btu/hr, the time required to reach venting pressure was found. The vent times for oxygen are shown in Figure 58. It is seen that for a 14-day no-loss standby the initial fill is 260 pounds, and 280 pounds for a 10-day no-loss standby. The total energy requirement for Configuration D' is 1358 kwh, of which 629 kwh are supplied by the SM cryogenics. Thus, the external device cryogenics must supply 738 kwh, which is equivalent to 533 pounds of oxygen. The total oxygen requirement for the power system is 981 pounds which, added to the ECS requirement of 450 pounds, gives a storage total of 1431 pounds. Of this amount, 640 pounds are stored in the SM, and therefore 971 pounds must be carried in the external device.

Figure 59 is a plot of the fluid remaining in the tank versus time for a Block II oxygen tank with an initial fill of 327 pounds. After 14 days, 284 pounds remain in the tank; the number of external device tanks required is then

$$\frac{791}{284} = 2.8$$

Therefore, three additional Block II oxygen tanks were placed in the external device.

Similar calculations for the venting characteristics of the Block II hydrogen tanks were made. Results are shown in Figure 60 for several heat-leak rates, and it is seen that there is no initial fill condition which gives a no-loss standby period greater than 9.7 days. Figure 61 shows the amount of fluid remaining in one Block II hydrogen tank versus time for different values of the initial fill weight.

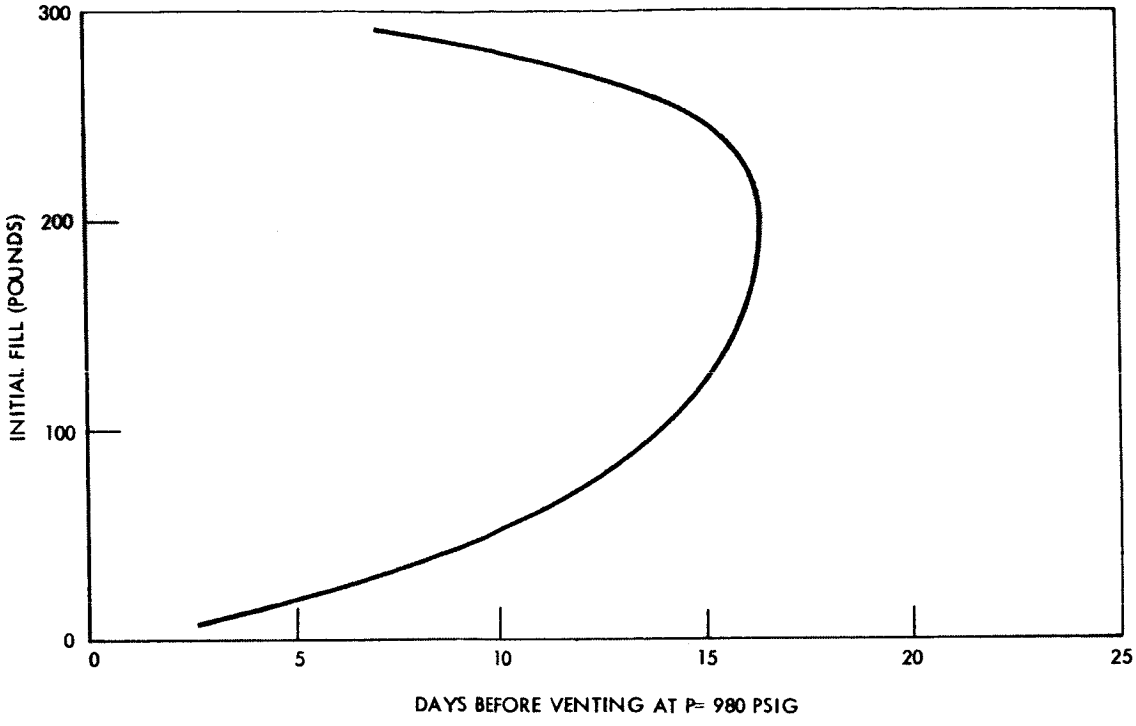

Figure 58. Standby Time Before Venting at 980 psig

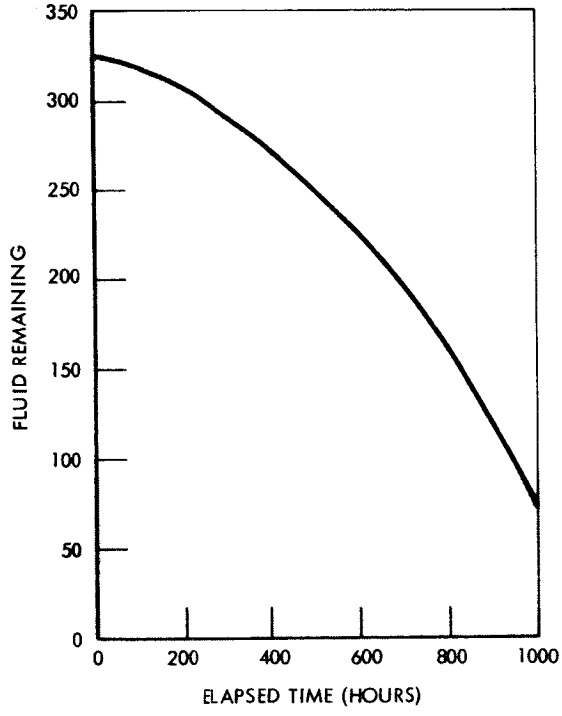

Figure 59. Block II O_2 Tank W_R Versus Time

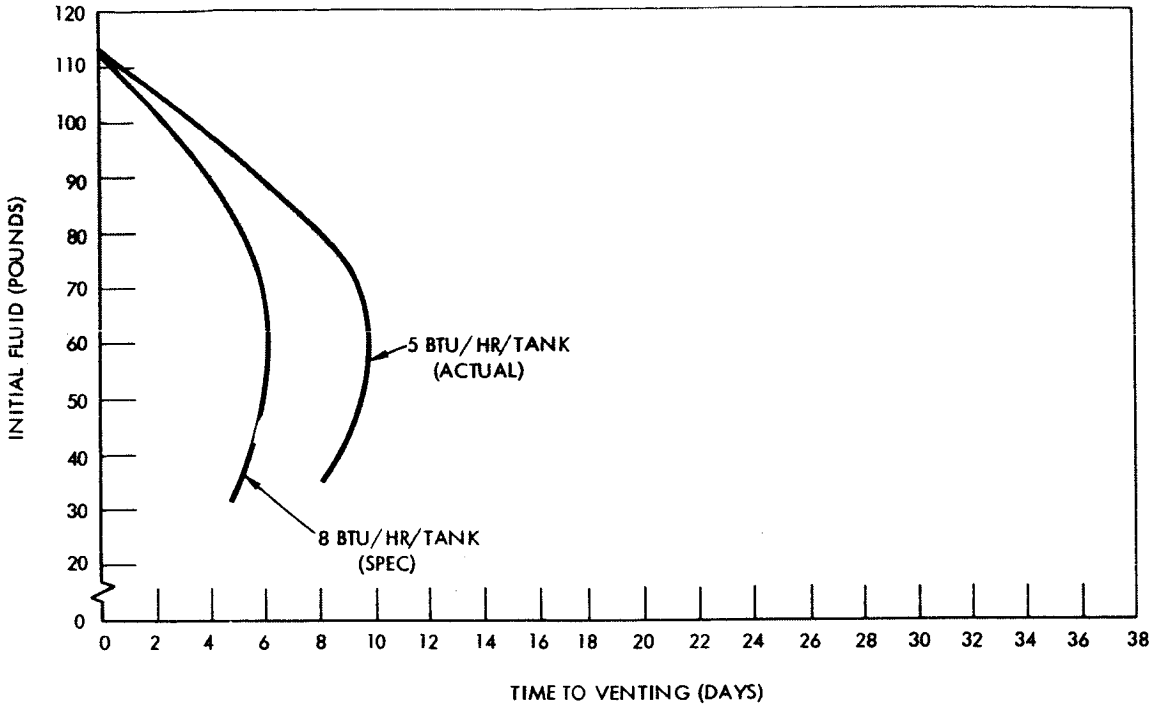

Figure 60. Storage of Cryogenic H_2 Without Venting (Block II Tanks)

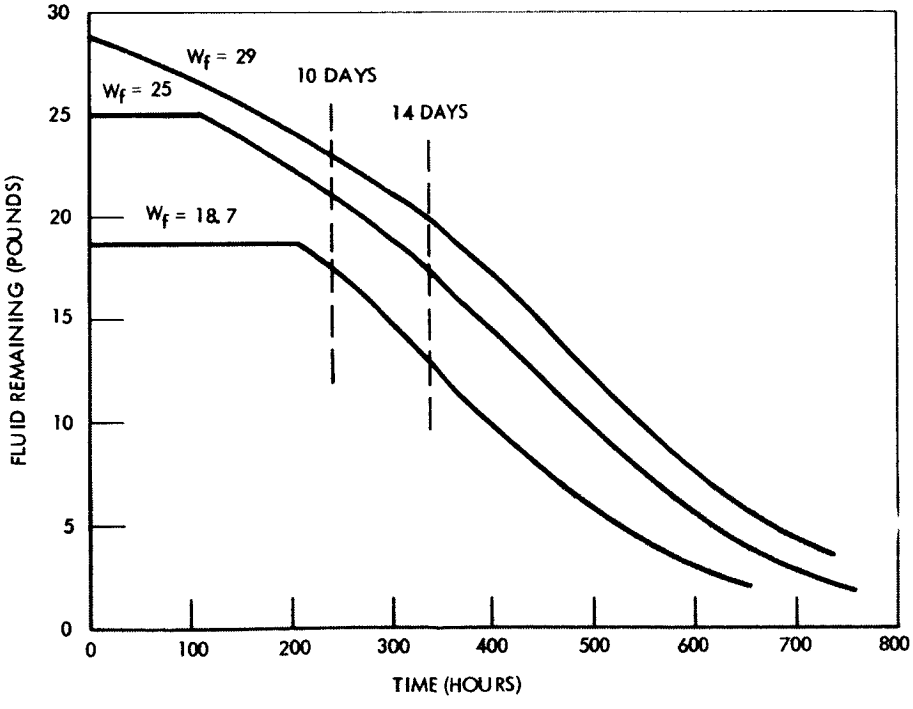

Figure 61. Block II H_2 Tank W_R Versus Time, Q_L = 5 Btu/Hour

The total hydrogen required for 1358 kwh is 123 pounds; 56 pounds (usable) are in the SM; therefore 67 pounds must be supplied from the external device. From Figure 61, with an initial fill of 29 pounds, 19.8 pounds (including residual) remain after 14 days. To obtain 67 pounds of usable reserve requires

$$\frac{67}{19.8 - 1.2} = 3.6, \text{ or 4 tanks}$$

After 10 days, 23 lb remain, requiring 3.08 tanks; since the calculations may be in error owing to the assumption of uniform internal temperature, four hydrogen tanks are installed in the external device. The hydrogen which is vented from the tanks during the first 10 to 14 days of the mission can be vented through the fuel cells, using the purge valves rather than depending upon a tank relief valve.

Originally, it had been intended to off-load the hydrogen tanks in order to delay the time at which venting initiates. The vented hydrogen from the external device would then be combined with the SM hydrogen, with the operating profile as shown in Figure 62. This not only would require hydrogen piping from the SM to the external device, but also is not workable with four tanks, there being less than 720 hours available operation.

Figure 63 shows the venting rates for four tanks, from heat-leak only, with initial fill weight as the parameter.

Effect of Use Rate

The Apollo Block II hydrogen and oxygen can be used to supply reactants for longer than 14-day missions provided that the use rate is always above the boiloff rate (caused by heat leaks) or provided that venting provisions are available to release excess tank pressure. Figures 64 and 65 show the maximum mission durations that can be obtained with various combinations of tanks at various loads. Where the boiloff rate exceeds the flow rate, the fluid is wasted by loading the fuel cells with a parasite load. From the curves, which are based on predicted values of heat leakage rather than the higher specification values, it is noted that there is a boil-off problem only in cases of three or more tanks used with two fuel cells. Thus, it is theoretically possible to achieve a 30-day mission with multiples of Block II tanks, provided the use rate and heat leaks are properly matched throughout the mission.

Alternative Solutions

Several alternative solutions to the D' case were studied to attempt to achieve a 30-day mission at the power loads of Flight 211. If four external device tanks are used, the usable weight per tank remaining after 14 days

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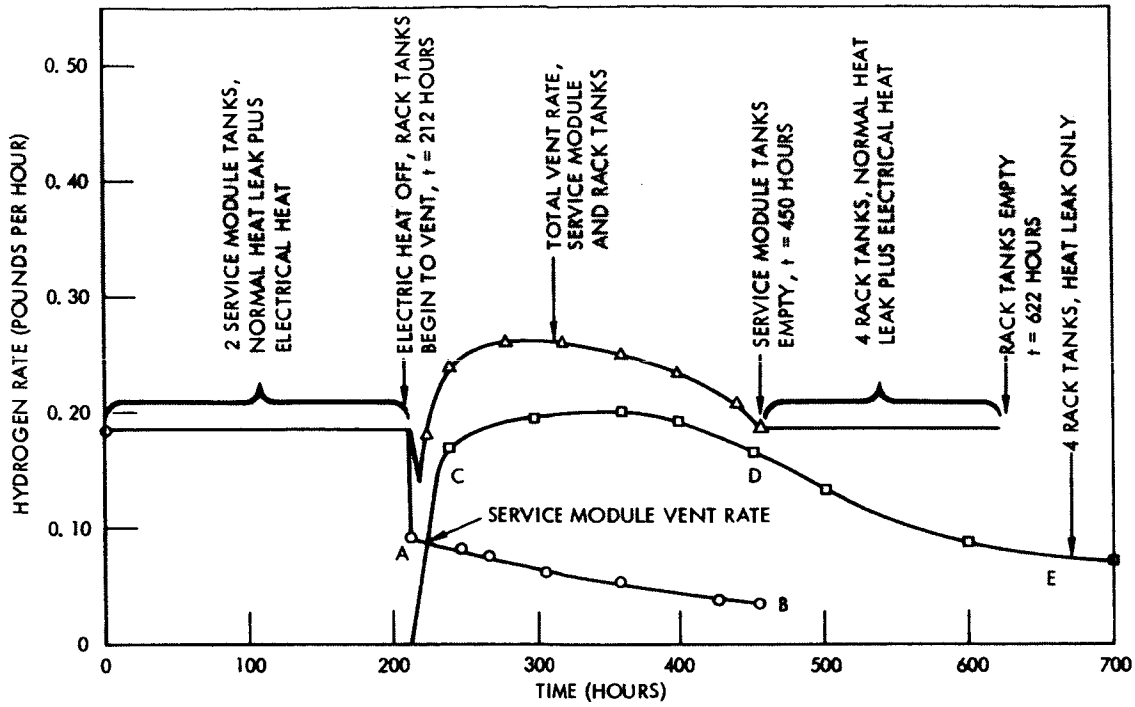

Figure 62. Operating Profile for Configuration D' with Rack H_2 Tanks Loaded to 18.7 Pounds

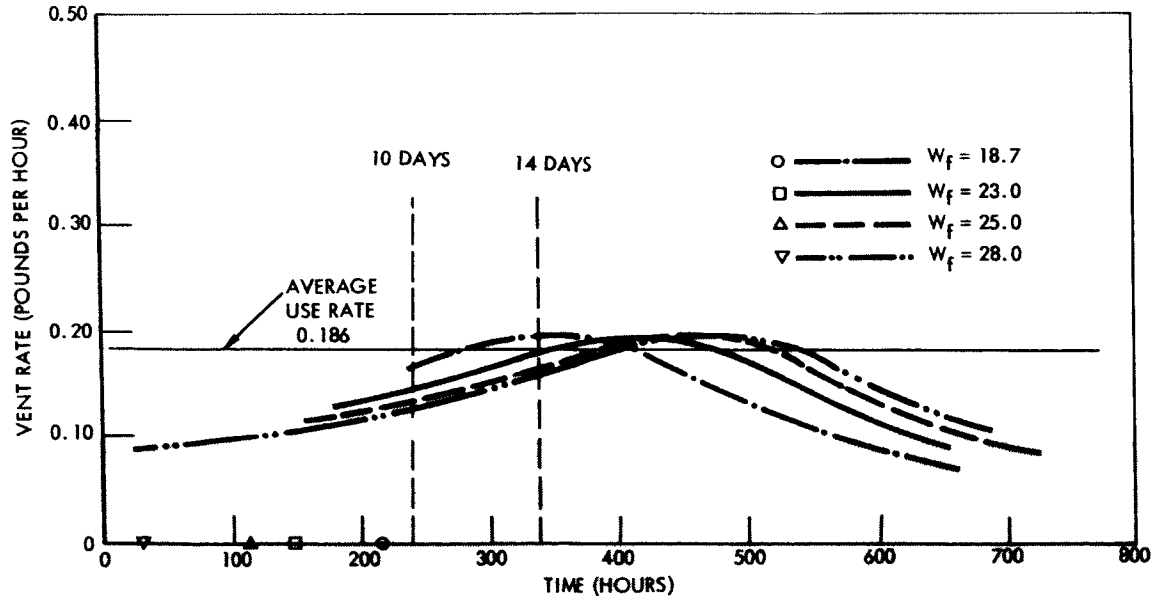

Figure 63. Venting Rate for Four Block II Hydrogen Tanks

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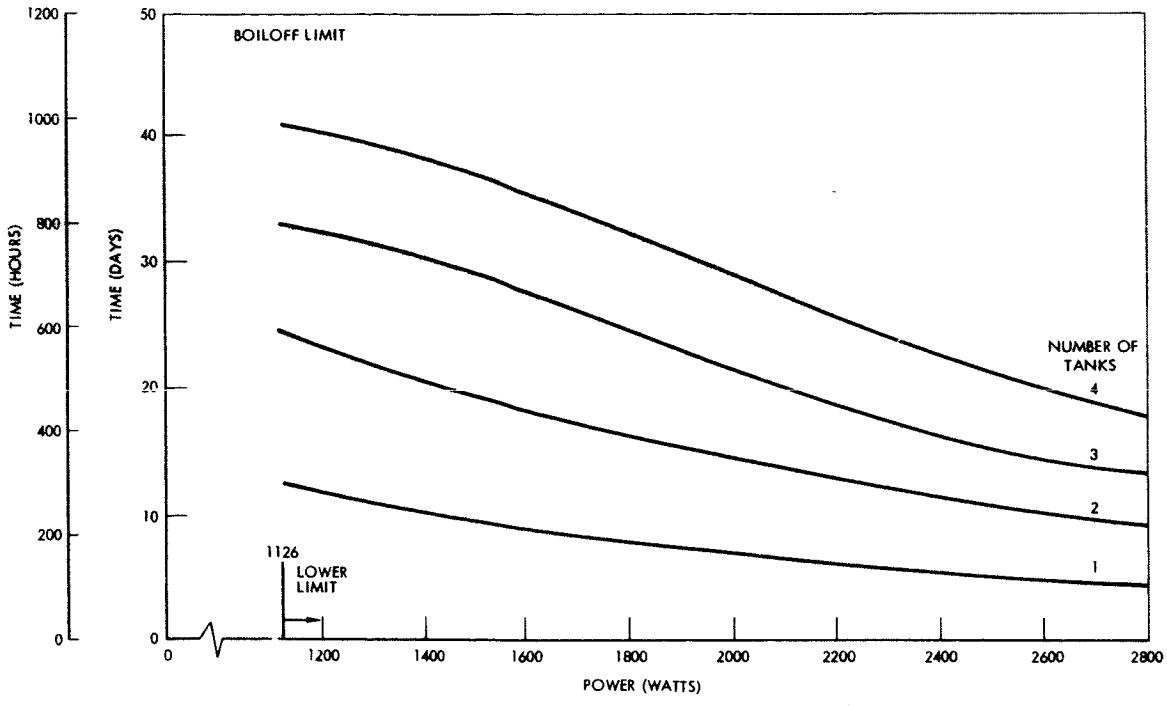


Figure 64. Use of Block II H_2 Tanks With Two Fuel Cells

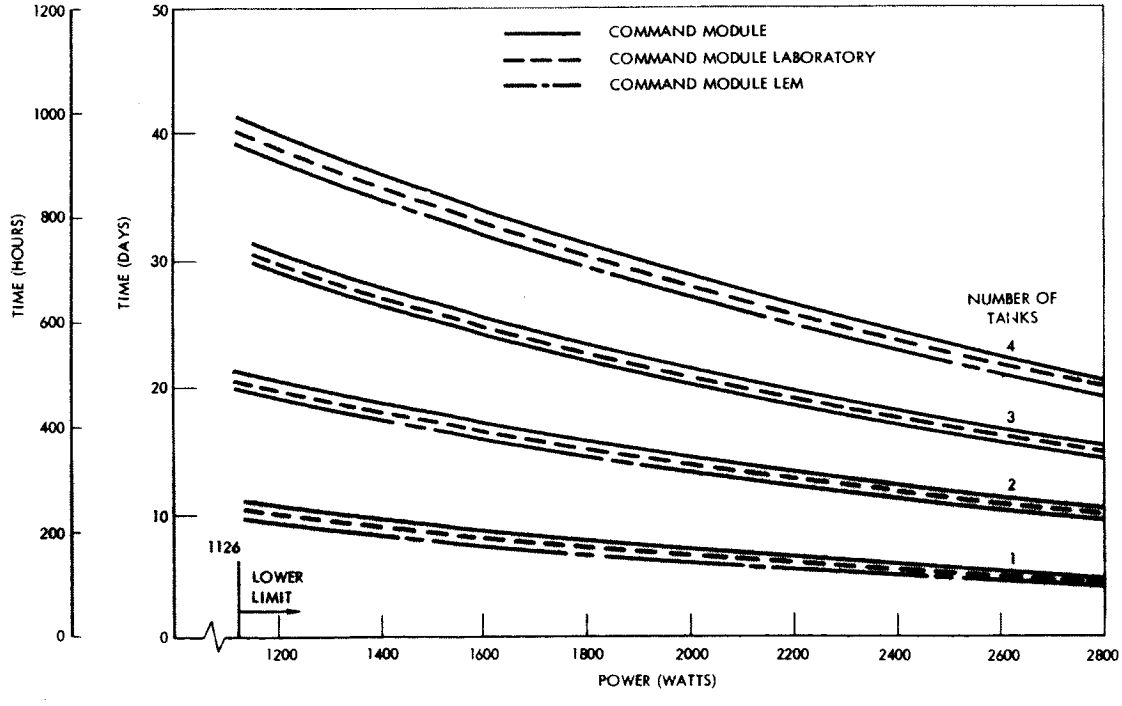


Figure 65. Use of Block II O_2 Tanks With Two Fuel Cells

is 18.6 pounds. The operating profile for this tank combination may be found in Figure 62. The SM tanks are used at the rate of 0.186 pounds per hour (0.093 pounds per each) from launch until 212 hours, at which time the external device tanks are fully pressurized and begin to vent. The electric heat to the service module tanks is shut off, and 9 pounds remaining in each tank, they follow the venting curve AB. The total vent fluid rate from the four external device tanks is given by curve CD, with the sum of the SM and external device vent rates shown immediately above. Note that the total vent rate (from heat-leak alone) exceeds the average flow requirement of 0.186. At 456 hours, the SM tanks are empty and the external device tanks vent from heat-leak only proceeds along DE. However, DE is below the required rate, so electric heat is added to bring up the rate. Under this condition, the external device tanks empty at 622 hours, or 26 days, which is short of the required 30 days.

Figure 66 shows an operating profile for two service module tanks and four external device tanks, with an initial fill of 23 pounds in each rack tank. For this case, the total operating time is 682, or 28.4 days.

Further analysis indicated that days' duration could only be achieved under these conditions by fully loading the external device tanks and venting fluid or wasting energy during early portions of the flights. Since the fuel

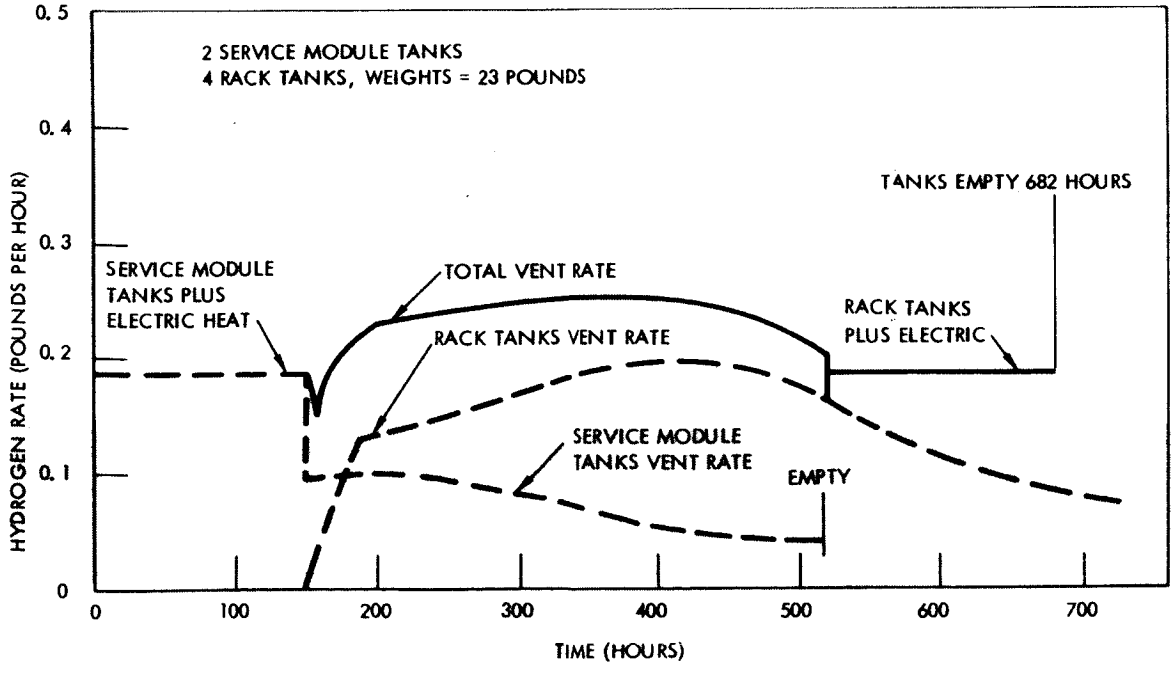


Figure 66. Operating Profile with Rack H_2 Tanks Loaded to 23 Pounds

cells in Configuration D' are life rated at only 400 hours, using operating fuel cells for venting purposes necessitates starting them early in the flight, thus adding to the life reliability problems. As discussed in the power systems section, there is also a problem in arranging for external device fuel cells to share the load with SM fuel cells so that they follow almost exactly the varying vent rates of the SM tanks and the external device tanks.

As the above discussion indicates, although there are analytical solutions, an entirely satisfactory solution to the D' case has not been found. One way to eliminate the entire D' cryogenic storage problem is to accelerate the development of the Configuration D hydrogen tank so that it is available for this flight. Discussions with Beech Aircraft indicate that this can be done, although the go-ahead date for this item would have to be moved up 4 months from the go-ahead of other subsystems to provide time allowances for a design error in the tank structure necessitating re-ordering of tank forgings after tank pressure tests.

CONFIGURATION D

This configuration uses a Block II SM with a mission duration of 45 days; this requires cryogenic tanks in the external device to supply hydrogen and oxygen after the SM tanks empty at 14 days. With the flight schedule for this configuration, the Apollo X tanks will be available and will therefore be used in the external device.

Table 32 shows that the maximum energy requirement for Configuration D is for Flight 229D, at 2446 kwh. Therefore, a capacity of 2450 kwh is used for all D flights, requiring rack power of 1830 kwh. This is equivalent to 166 pounds of hydrogen. Based on an analysis similar to that for Configuration D', Figure 66 shows that an 83-pound load in an Apollo X tank has a no-loss standby time of 14 days. Therefore, two Apollo X tanks with a capacity of 104 pounds each are off-loaded to 83 pounds each, and would provide the hydrogen required. Similarly, it was determined that two Apollo X oxygen tanks on the external device will provide sufficient capacity.

CONCLUSIONS

The above study points out that the presently available Block II cryogenic storage system and the proposed Apollo X system are suitable for the requirements of Configuration 1, C, and D. As explained, there is presently no clear-cut solution for Configuration D'; however, it is estimated that the mission can be accomplished by using 4 fully-loaded Block II hydrogen tanks and 3 Block II oxygen tanks in the external device. It is evident that further study is required to analyze fully the D' requirements and the ultimate capability of the Block II system.

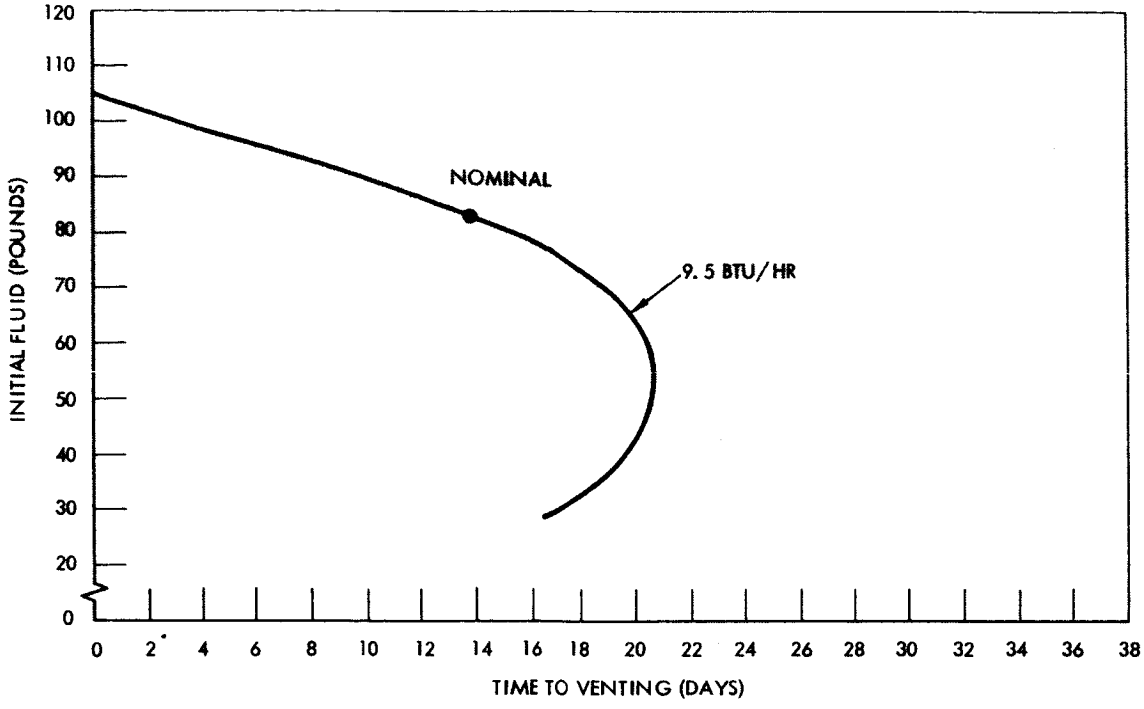


Figure 67. Storage of Cryogenic H_2 Without Venting (Apollo X Tanks)

ENVIRONMENTAL CONTROL AND LIFE SUPPORT SYSTEMS

This section is divided into three parts. The first part presents the results of thermal analyses concerning the pallet, rack, command module, and parachute compartment which were used to analyze environmental control requirements; the second part considers the environmental control system; the third part discusses the characteristics of the life support system.

THERMAL ANALYSIS

During the orbital phase of the missions, the vehicle will be subjected to external heating from the sun, both direct and reflected energy, and the earth. For the purposes of this study, these heating factors were computed by an IBM 7094 digital computer program (a modification of the program described in ASD TN 61-83) which computes the incident heat loads on a plane surface as a function of the orbital position (true anomaly).

Since the program requires that the vehicle be simulated by means of plane surfaces (maximum of 12) the command module was simulated by a truncated octagonal pyramid; the rack, or service module, was simulated by an octagonal prism. Surface nomenclature and the location of the vehicle axes are illustrated in Figure 68.

A total of 38 computer runs were made to determine the incident heat loads on the command module and rack for the various orbits and orientations. These are described in Table 37. For all runs, a solar constant of 443 Btu/ ft^2/hr and an Earth's albedo of 0.4 were used. Values of surface emissitivity of 0.9, and solar absorbtivity of 0.2 were used to determine the absorbed heat for all but run 1 (which used $\epsilon = .88$, $\alpha_s = 0.14$). For the thermal analysis of cases with other surface properties, the program output of incident heat flux was used.

Typical graphic output from the program is illustrated in Figure 69, and a comparison of absorbed heat load for various orbits is given in Figure 70.

PALLET THERMAL ANALYSIS

A simplified thermal analysis was made of the experimental payload to be installed in Bay 1 of the service module for Flight 518. This payload

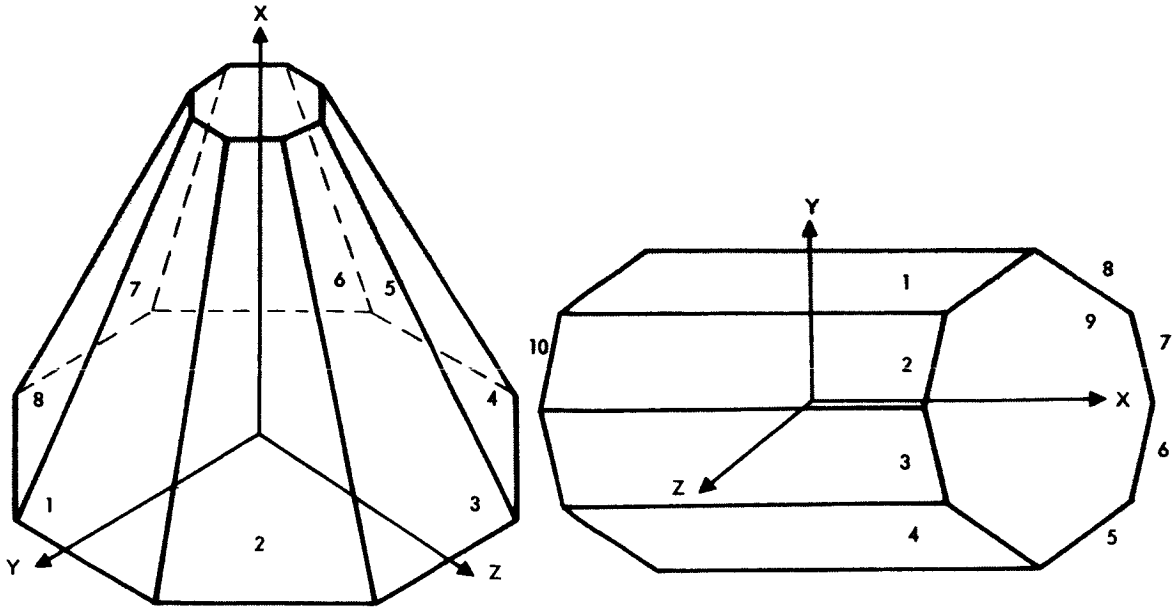


Figure 68. Vehicle Surface and Axes Designation

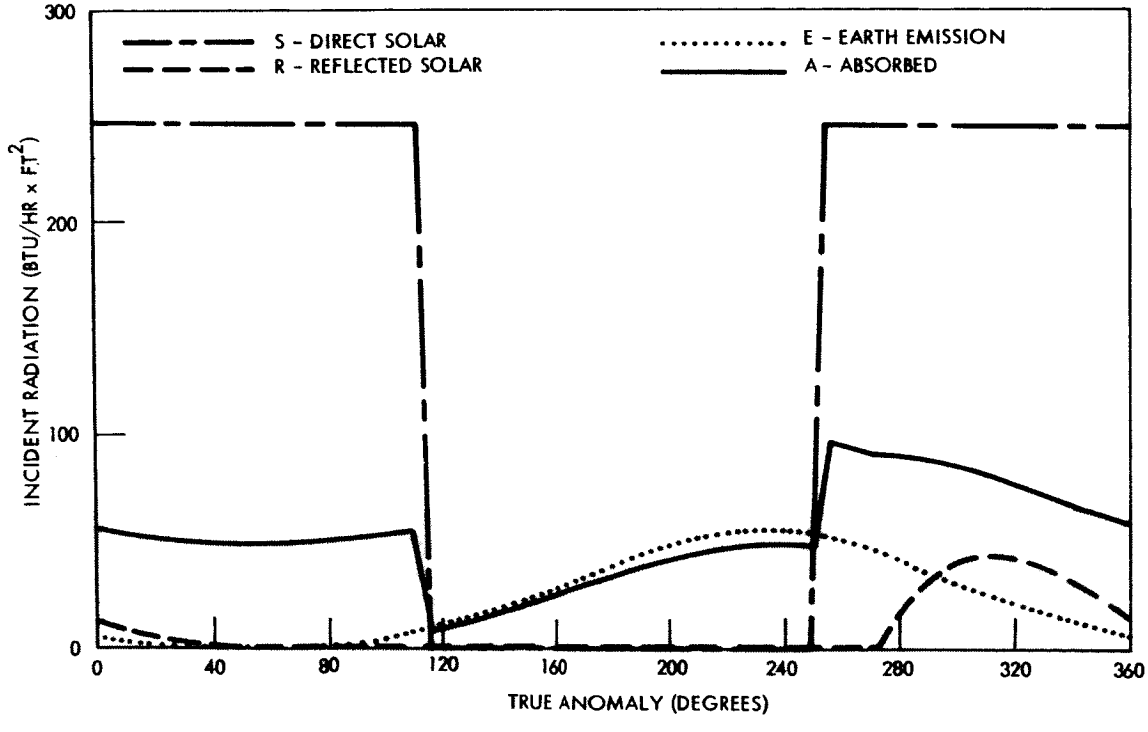


Figure 69. Typical Program Output Command Module, Surface 1 (Run 3)

Table 37. Description of Orbital Heating Cases

Run	Module	Orbital Elements					Orientation
		H	I	T	Ω	ECC	
1	CM	200	28.5	80	0	0	Y axis toward Earth
2	CM	200	28.5	80	0	0	Y axis toward Earth
3	CM	200	28.5	80	0	0	X axis toward east (Inertial)
4	CM	200	90.	314	75	0	Y axis toward Earth
5	CM	200	90.	314	75	0	X axis east
6	CM	200	50.	80	0	0	Y axis toward Earth
7	SM	200	28.5	80	0	0	X axis toward Earth
8	SM	19357	0.	222	0	0	X axis toward Earth
9	CM	200	50.	80	0	0	X axis east
10	CM	200	81.5	80	0	0	X axis east
11	CM	200	81.5	80	0	0	Y axis toward Earth
12	CM	200	99	284	0	0	X axis toward west
13	CM	200	99.	284	0	0	Y axis toward Earth
14	CM	19357	0.	222	0	0	Y axis toward Earth
15	CM	19357	0.	222	0	0	Y axis toward Sun
16	CM	700	50.	80	0	0.065	Y axis toward Sun
17	CM	700	50.	80	0	0.065	Y axis toward Earth
18	CM	200	30.	80	0	0	Y axis toward Sun
19	CM	200	50.	80	0	0	Y axis toward Sun
20	SM	200	50.	80	0	0	Y axis toward Earth
21	SM	200	50.	80	0	0	Y axis toward Sun
22	SM	200	50.	80	0	0	X axis toward Sun
23	SM	200	28.5	80	0	0	Y axis toward Sun
24	SM	200	28.5	80	0	0	X axis toward Sun
25	SM	200	90.	314	75	0	Y axis toward Earth
26	SM	200	99.	284	0	0	X axis toward Sun
27	SM	200	99.	284	0	0	Y axis toward Earth
28	SM	200	81.5	80	0	0	Y axis toward Earth
29	SM	200	81.5	80	0	0	X axis toward east
30	SM	200	30.	80	0	0	Y axis toward Sun
31	SM	200	30	80	0	0	Y axis toward Earth
32	SM	19357	0.	222	0	0	Y axis toward Sun
33	SM	19357	0.	222	0	0	X axis toward east
34	SM	700	50.	80.	0	0.065	Y axis toward Earth
35	SM	700	50.	80	0	0.065	Y axis toward Sun
36	SM	200	90.	314	75	0	X axis east
37	SM	200	28.5	80	0	0	X axis east
38	SM	19357	0	222	0	0	Y axis toward Earth

H = Altitude of apogee, nmi
I = Inclination, degrees
T = Epoch, days
Ω = Right ascension of ascending node, degrees
Ecc = Eccentricity

consists of radar and associated electronic equipment. The analysis was performed to determine if the temperature would remain within the limits of 32 to 150 F.

The payload was simulated by a single active node network as shown in Figure 71. The weight of the payload was given as 525 pounds; it was assumed that the structure, environmental control equipment, and batteries would weight an additional 500 pounds. An average specific heat of 0.2 was assumed for all equipment. The internal heat loads were given as 50 watts during standby and 200 watts during operation, with peak loads of 2 kw.

Three computer runs were made to determine the payload temperatures with different surface coatings and insulation values. Because the exact operating profile was unknown, a constant internal heat of 200 watts (683 Btu/hr) was used for the runs; two other assumptions were made -- that the peak load of 2 kw occurred during transmission, and that the combination of pulse width and PRF was such that the transmitter's contribution to the internal load was included in the 200 watt figure.

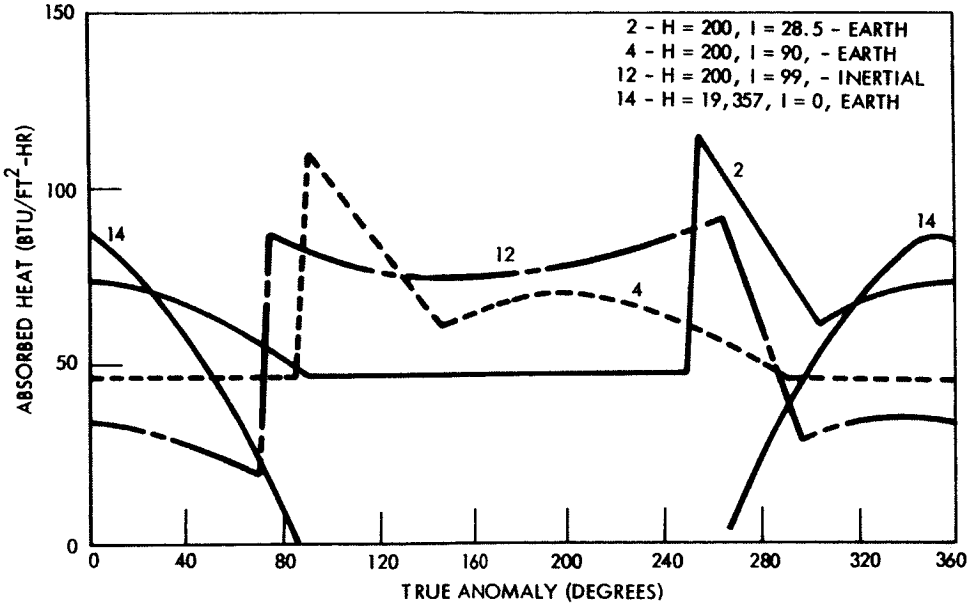


Figure 70. Absorbed Heat by Surface 1 of Command Module for Various Orbits

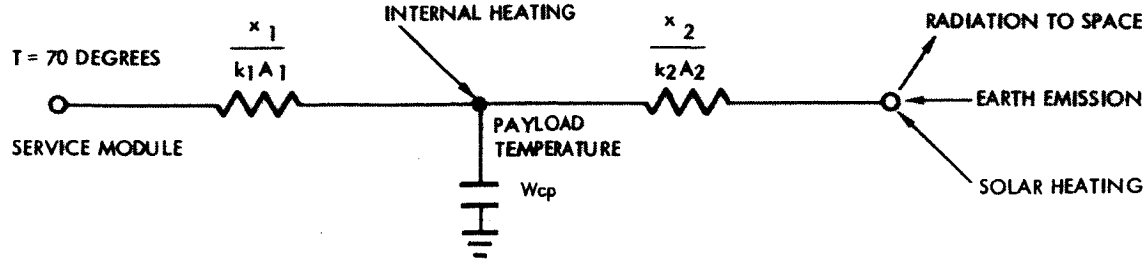


Figure 71. Pallet Thermal Network

The temperature of the payload at the end of each orbit, for 16 orbits is shown in Figure 72. Temperatures had not stabilized at the end of the run, but they were sufficiently close to indicate that a trend has been established. Figure 72 shows that, with the assumptions made, it will be possible to maintain the payload temperature within the desired limits. However, it will be necessary to repeat the analysis using a more detailed network to determine the temperature distribution within the payload.

RACK THERMAL ANALYSIS

Many possible experimental equipment configurations will be installed in the rack. Additionally, a number of different orbital situations will be encountered. As a consequence, the rack and its experimental packages will cover a wide range of external environmental conditions and internal heat generation rates. However, it is not the purpose of this study to examine each of the situations in detail. Therefore, this study's basic approach is the general examination of whether the experiments rack will require heating, cooling, or both. This necessitated a choice of orbital conditions that would provide the maximum and minimum external heating rates. The following missions and orientations were used:

1. Earth-oriented, 200-nmi altidude, 30 degree inclination.

2. Sun-orientated, 200-nmi altitude, 30 degree inclination.

3. Synchronous Earth orbit, 19,357-nmi altitude, 0 degree inclination.

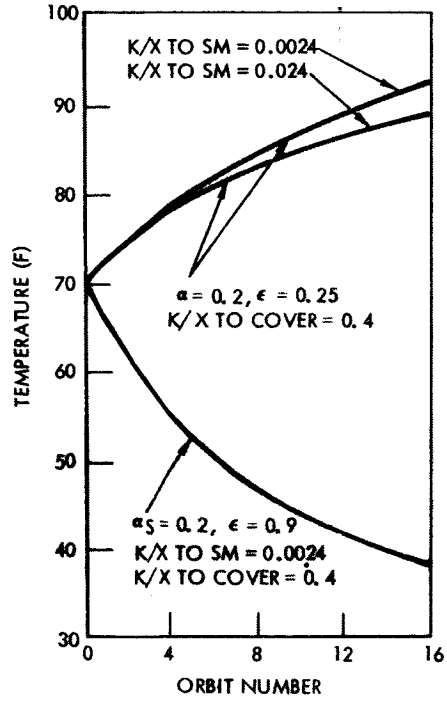


Figure 72. Pallet Payload Temperature (at End of Orbit)

To determine rack temperature variation and the required heating and/or cooling necessary to maintain specified air lock and experiment temperatures, a series of temperature-time histories was calculated. This series was based on the orbital conditions stated above. For all cases, the following assumptions were made:

1. The temperature of the air lock atmosphere will be maintained at 75 F.

2. The temperature of the various experiments outside the airlock will be maintained between 65 and 75 F. For the purposes of this study, it was assumed that the experiment temperatures will be the same as the shelves upon which they are mounted.

3. Any insulation required for thermal protection will be located on the outer skin of the rack. It is realized that in actuality it may be better to insulate either the airlock or the experiments -- or perhaps only certain sectors of the rack. However, due to the undefined nature of the many possible missions and experiments, it was necessary to make this assumption to conduct the study on a generalized basis.

4. Each sector contains two shelves.

5. Each fuel cell, when installed, will leak approximately 100 watts (342 Btu/hr) to the adjacent rack structure. This assumes that the fuel cells will be well insulated from the rack structure.

6. Heat transfer effects by conduction and radiation between the rack and the command module were assumed to be negligible.

7. Although all analyses were performed on a transient basis, this assumption was based upon the weight and specific heat of the rack structure, (i.e., experiment weights were not included).

The analog networks used to perform the study are shown in Figures 73 and 74. A total of 16 IBM runs were made using the same basic networks, with varying orbital inputs and insulation efficiencies for the different missions. A significant result of these analyses was the fact that in none of the situations studied did there appear to be a need for rack space radiators -- i.e., there is a net heat loss per orbit to the environment.

It should be noted that this heat loss resulted from controlling the airlock and other rack areas to specific temperatures; the effects of internal heat loads generated by personnel or equipment -- other than the fuel cells -- were not included. With minimum insulation, the heat loss

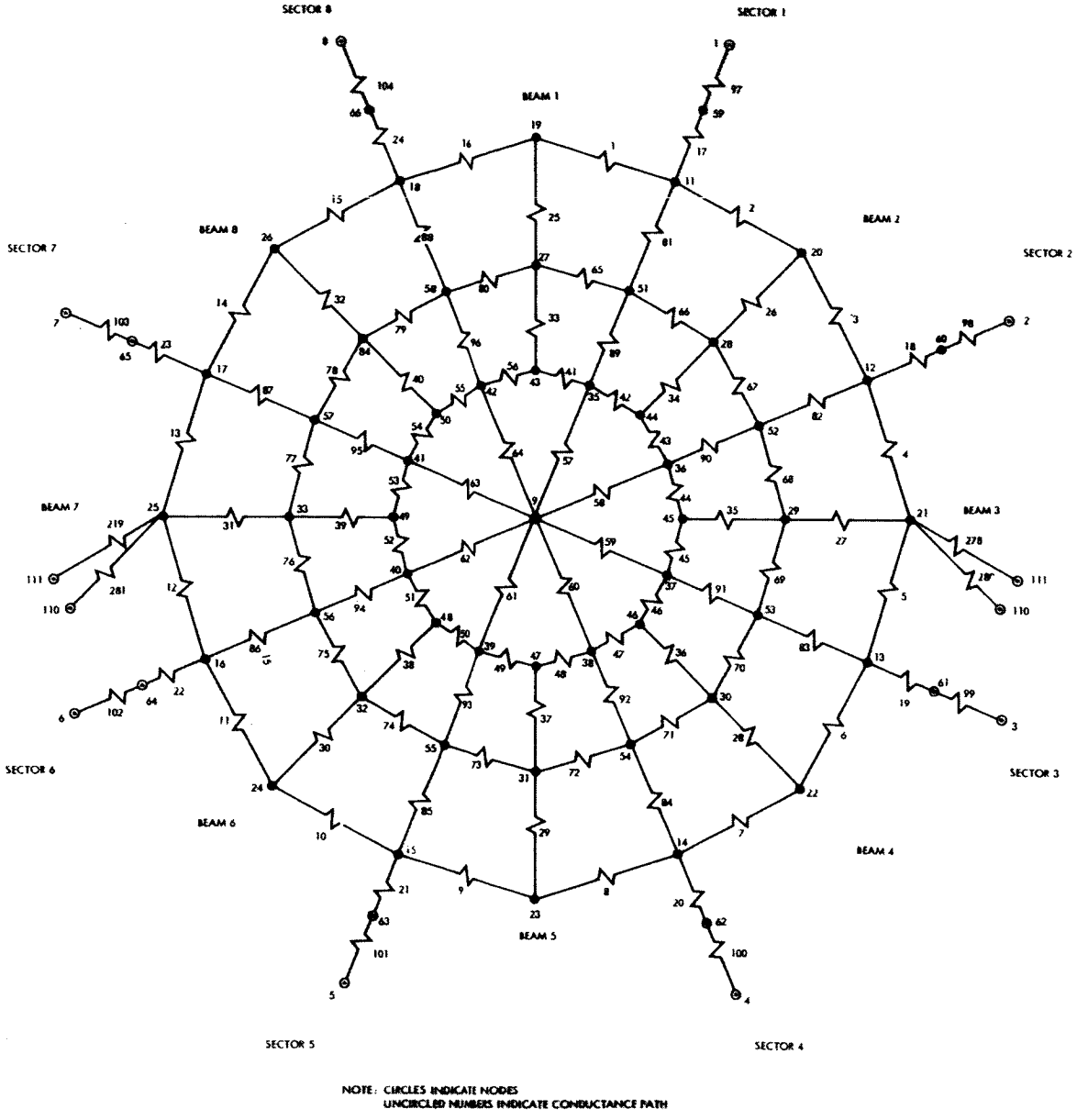


Figure 73. Primary Rack Thermal Network

from the rack as a whole will be considerably more than that generated internally. Thus, it will be possible to utilize internally generated heat to maintain specified temperatures, providing the proper insulation is used to control the heat leak to space.

Low-Earth Orbit, Earth-Oriented

This case involved a 200-nmi Earth orbit with the x-axis tangential to the orbit path. A total of six runs was made, each for ten consecutive orbits, with an orbit period of 1.53 hours. These runs were for the following cases:

1. Insulation k/x = 0.18

   Solar absorptivity (α_s) = 0.2
   Infrared emissivity (ϵ) = 0.9
   Airlock controlled to 75 F
   Shelves controlled to 65-75 F
   Two continuously operating fuel cells (200 w)

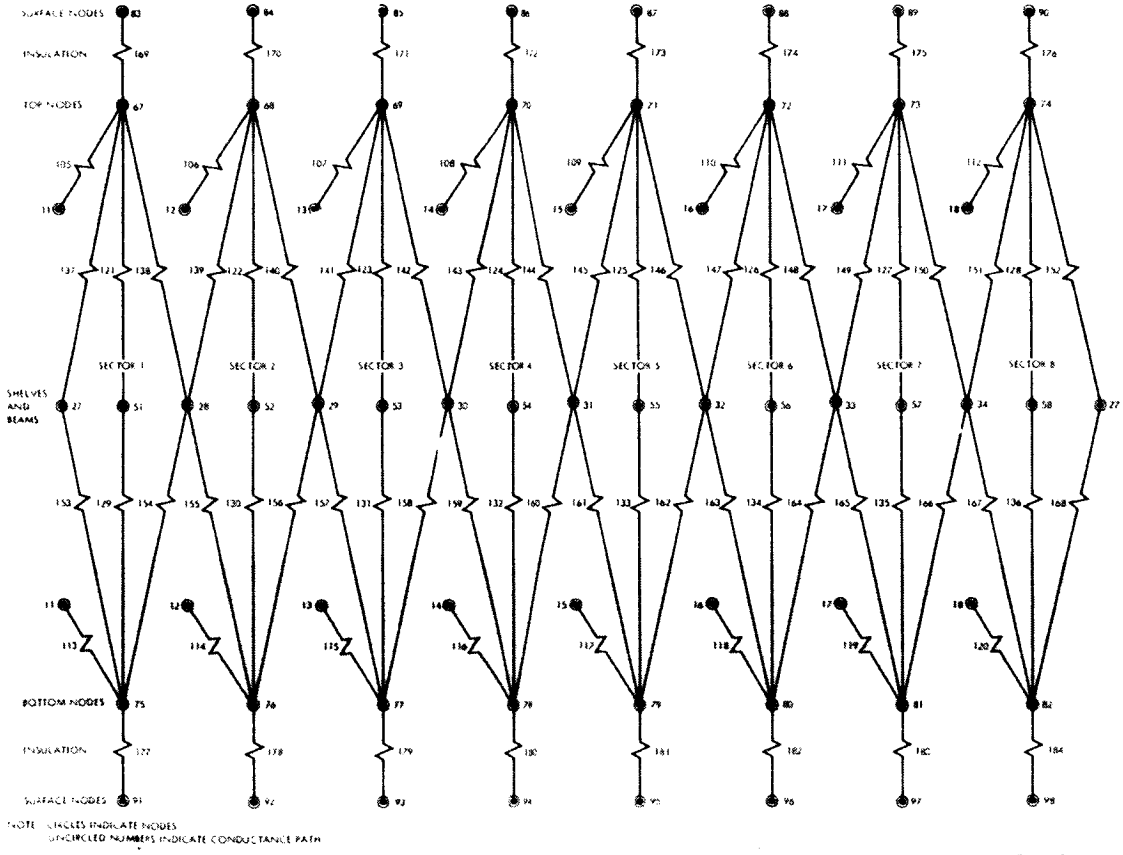


Figure 74. Rack Thermal Network, Internal Radiation and Insulation

2. $k/x = 0.18$, $\alpha_s = 0.2$, $\epsilon = 0.25$

   Air lock and shelves temperature controlled
   Two fuel cells

3. $k/x = 0.09$, $\alpha_s = 0.2$, $\epsilon = 0.25$

   Air lock and shelves temperature controlled
   Two fuel cells

4. $k/x = 0.018$, $\alpha_s = 0.2$, $\epsilon = 0.25$

   Air lock and shelves temperature controlled
   Two fuel cells

5. $k/x = 0.018$, $\alpha_s = 0.2$, $\epsilon = 0.25$

   Air lock temperature controlled, but not shelves
   Two fuel cells

6. $k/x = 0.018$, $\alpha_s = 0.2$, $\epsilon = 0.25$

   Air lock temperature controlled, but not shelves
   No fuel cells

Total heat loss as a function of orbit time is presented in Figures 75 and 76. Heat loss from the airlock only is presented in Figures 77 and 78. Rack surface temperatures are presented in Figure 79. It can be seen that thermal surface properties are a significant factor in heat loss control. Case 1, which utilizes a white solar reflecting finish (α_s) = 0.2, ϵ = 0.9), loses an undesirably large amount of heat. Case 2, which utilizes a low emissivity coating with an α_s/ϵ ratio of less than unity, loses far less heat with the same insulation effectiveness. Coatings such as used for case 2 are available and have been used on several satellites, notably the aluminum powder-silicone coating used on the Orbiting Solar Observatory[1].

It can be seen that the overall loss of heat can be controlled as desired by varying the surface coating and insulation effectiveness. The insulation effectiveness will depend to a great extent upon the amount of heat generated by the experiments; therefore, each mission will probably present a unique case. The insulation assumed for cases 1 and 2 consisted of 1.0

[1] Neel, Carr B., Research on the Stability of Thermal Control Coatings for Spacecraft, NASA TMX-51, 196, Ames Research Center, Moffett Field, California, October 1963.

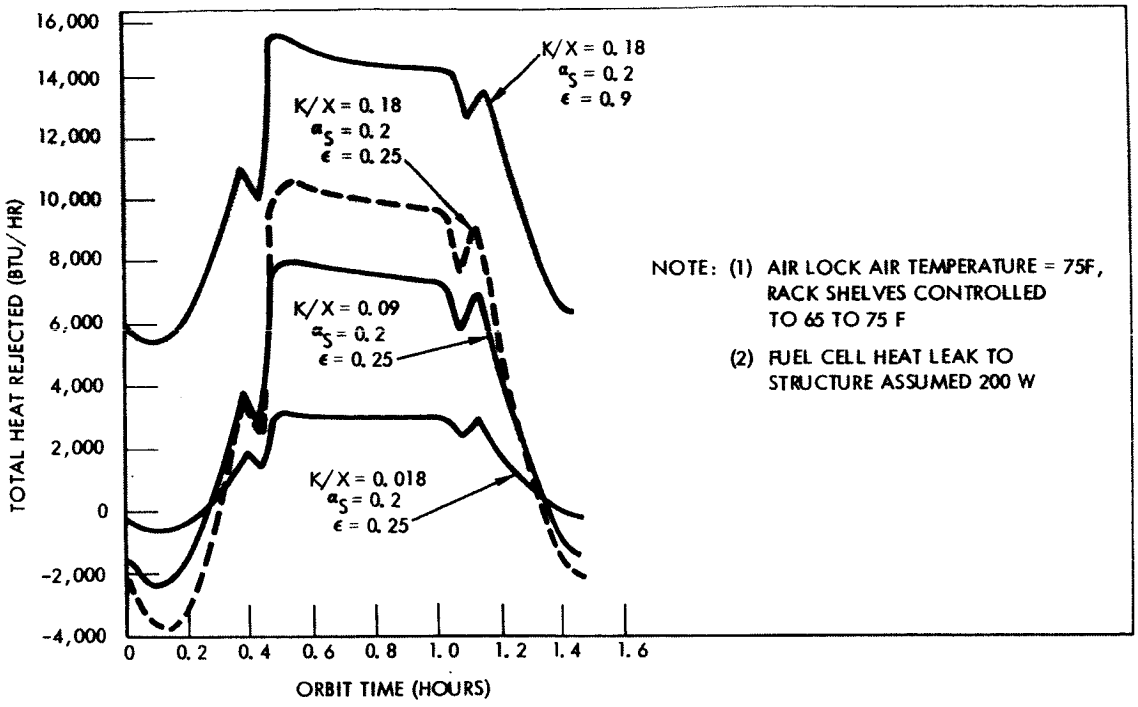


Figure 75. Rack Heat Loss for 200-Nautical-Mile, Earth-Oriented Orbit, for Various Insulation Efficiencies and Thermal Surface Properties

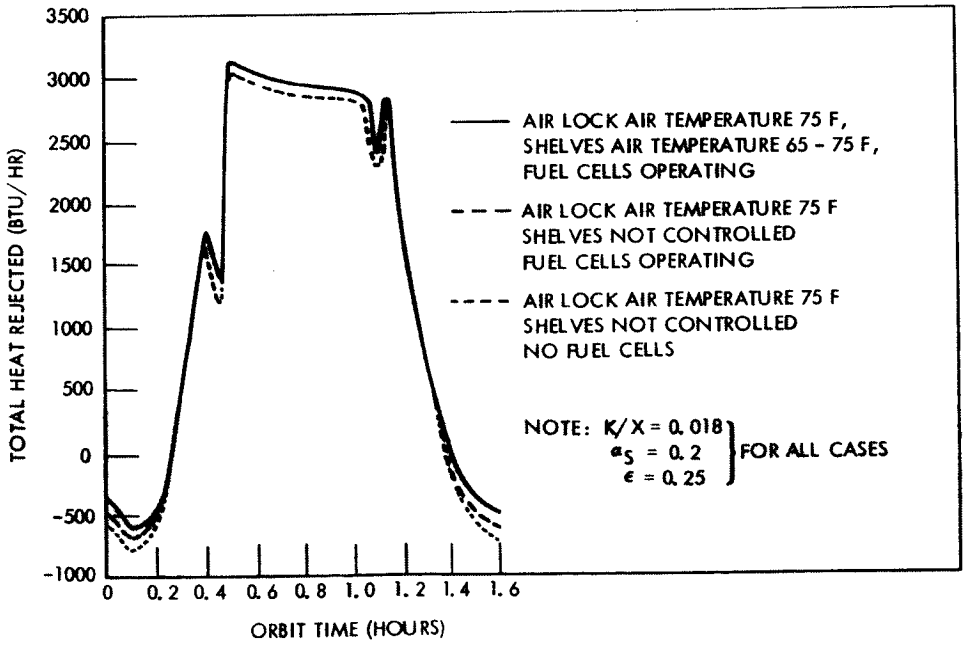

3500
3000
2500
2000
1500
1000
500
0
-500
-1000
TOTAL HEAT REJECTED (BTU/HR)
0 0.2 0.4 0.6 0.8 1.0 1.2 1.4 1.6
ORBIT TIME (HOURS)
AIR LOCK AIR TEMPERATURE 75 F, SHELVES AIR TEMPERATURE 65 - 75 F, FUEL CELLS OPERATING
AIR LOCK AIR TEMPERATURE 75 F SHELVES NOT CONTROLLED FUEL CELLS OPERATING
AIR LOCK AIR TEMPERATURE 75 F SHELVES NOT CONTROLLED NO FUEL CELLS
NOTE: K/X = 0.018, α_S = 0.2, ϵ = 0.25 } FOR ALL CASES

Figure 76. Rack Heat Loss for 200-Nautical-Mile, Earth-Oriented Orbit, for Various Internal Heat Loads

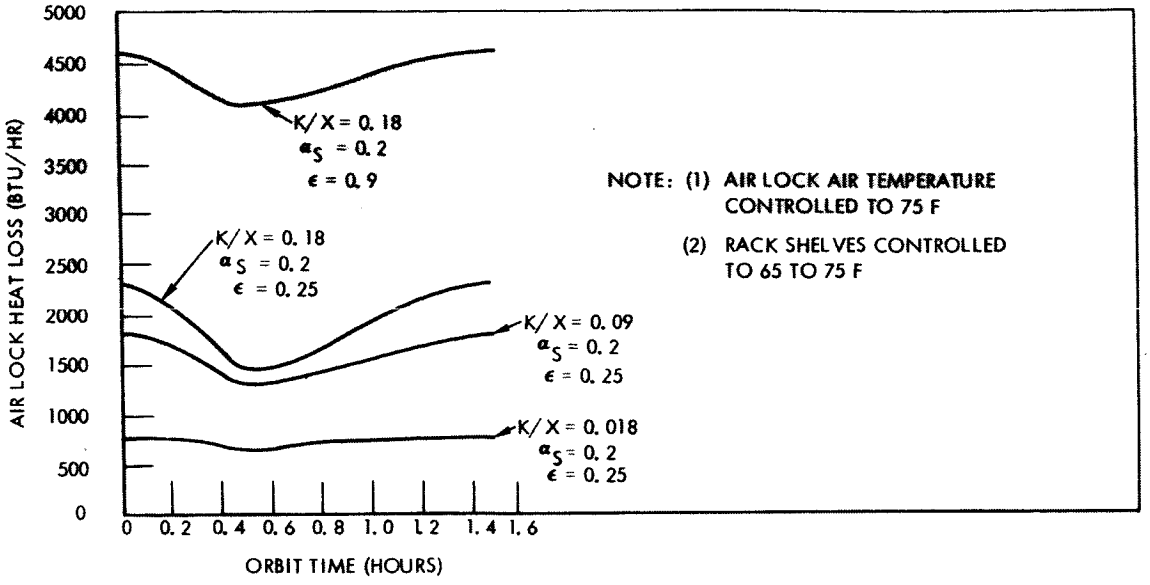


Figure 77. Air Lock Heat Loss for 200-Nautical-Mile, Earth-Oriented Orbit, for Various Insulation Efficiencies and Thermal Surface Properties

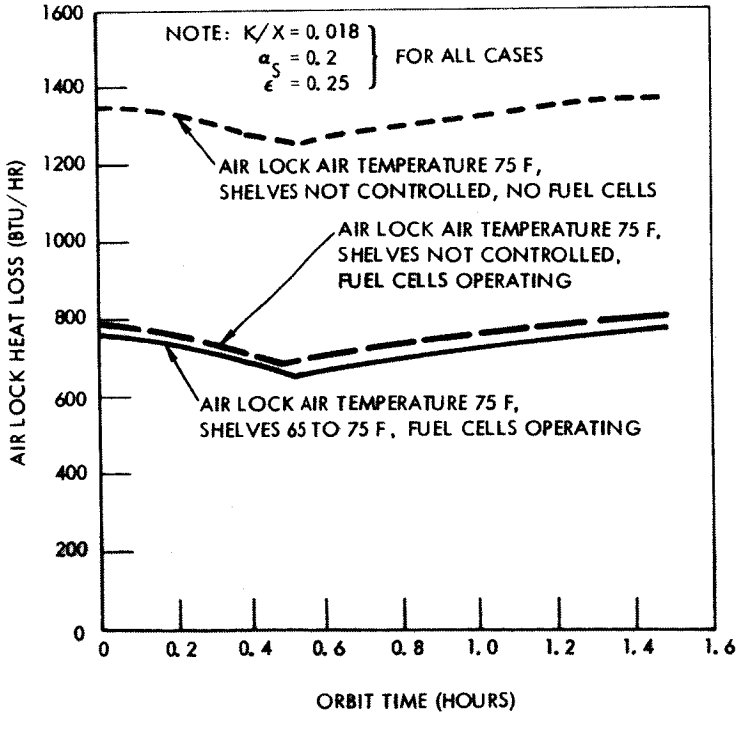

AIR LOCK HEAT LOSS (BTU/HR)
1600
1400
1200
1000
800
600
400
200
0
NOTE: K/X = 0.018
α_S = 0.2
ϵ = 0.25
FOR ALL CASES
AIR LOCK AIR TEMPERATURE 75 F, SHELVES NOT CONTROLLED, NO FUEL CELLS
AIR LOCK AIR TEMPERATURE 75 F, SHELVES NOT CONTROLLED, FUEL CELLS OPERATING
AIR LOCK AIR TEMPERATURE 75 F, SHELVES 65 TO 75 F, FUEL CELLS OPERATING
0 0.2 0.4 0.6 0.8 1.0 1.2 1.4 1.6
ORBIT TIME (HOURS)

Figure 78. Air Lock Heat Loss for 200-Nautical-Mile, Earth-Oriented Orbit, for Various Internal Heat Loads

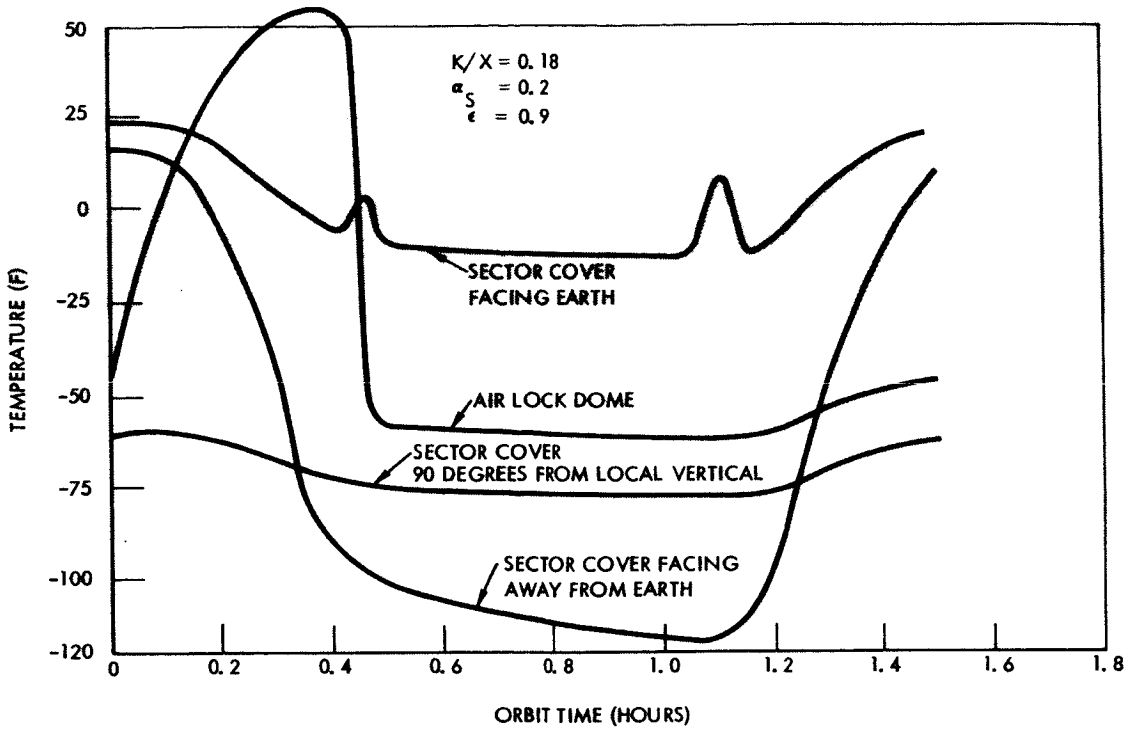


Figure 79. Rack Outside Skin Temperatures for 200-Nautical-Mile, Earth-Oriented Orbit

inch of polyurethane foam with an average thermal conductivity of 0.015 Btu/(hr)(ft^2)(°F/ft), which resulted in a k/x value of 0.18. Case 3 used double this strength, i.e., k/x = 0.09. Cases 4, 5, 6 utilized an insulation better than case 1 by a factor of 10. In order to achieve this very low figure, it will probably be necessary to use a multiple-layer superinsulation, such as Linde SI-62 or NRC-2, with an average effective (installed) thermal conductivity on the order of 10 x 10^{-4} Btu/(hr)(ft^2)(°F/ft).

Low-Earth Orbit, Solar-Oriented

This is also a 200-nmi Earth orbit; however, for this orbit the vehicle is solar oriented so that the rack continually faces the sun (positive x-axis towards sun). A total of three runs were made, corresponding to cases 2, 3, and 4 above:

1. k/x = 0.18, α_s = 0.2, ϵ = 0.25

   Air lock and shelves temperature controlled
   Two fuel cells continuously operating

2. $k/x = 0.09$, $\alpha_s = 0.2$, $\epsilon = 0.25$

   Air lock and shelves temperature controlled
   Two fuel cells, continuously operating

3. $k/x = 0.018$, $\alpha_s = 0.2$, $\epsilon = 0.25$

   Air lock and shelves temperature controlled
   Two fuel cells continuously operating

The results are presented in Figures 80 and 81. The heat losses are quite comparable to those obtained for the Earth-oriented cases. Total heat loss from the rack is somewhat less than before, but heat loss from the airlock is somewhat greater than before. This latter phenomenon is probably due to the fact that the air lock dome is directly facing the sun for longer periods.

Synchronous Orbit, Solar-Oriented

This is a 19,300-nmi altitude synchronous orbit, with the rack oriented towards or away from the sun. A total of seven runs were made for this situation. Because Earth thermal effects are negligible for such an orbit, the heat loss rates reach an essentially constant value -- i.e., no appreciable variation during orbit. All cases were run assuming surface finish properties of $\alpha_c = 0.2$, $\epsilon = 0.25$. The results are summarized in Table 37. It can be seen immediately that there is a significant difference in results between cases 1, 2, and 3, where the rack is pointed directly towards the sun -- and cases 4, 5, 6, where the rack is pointed away from the sun and shielded by the rest of the vehicle. In the latter cases, the rack sees essentially nothing except black space and thus will reject a maximum amount of heat.

Case 7 is identical to Case 4 in that it is oriented away from the sun. In this case, however, only the air lock is maintained at a controlled temperature; the rest of the rack is allowed to seek its own level. It is apparent that the rack as a whole loses less heat than before, but that all this heat (7205 Btu/hr) must come from the air lock. This is by far the largest heat loss from the air lock of all the cases studied; an obvious solution would be to insulate the air lock from the rest of the rack.

Conclusions

It is apparent from the analyses discussed above that a rack ECS radiator probably will not be needed. However, a liquid-loop temperature control system will be probably necessary to transport heat from local hot spots and deliver heat to areas that have a tendency to cool. In this manner,

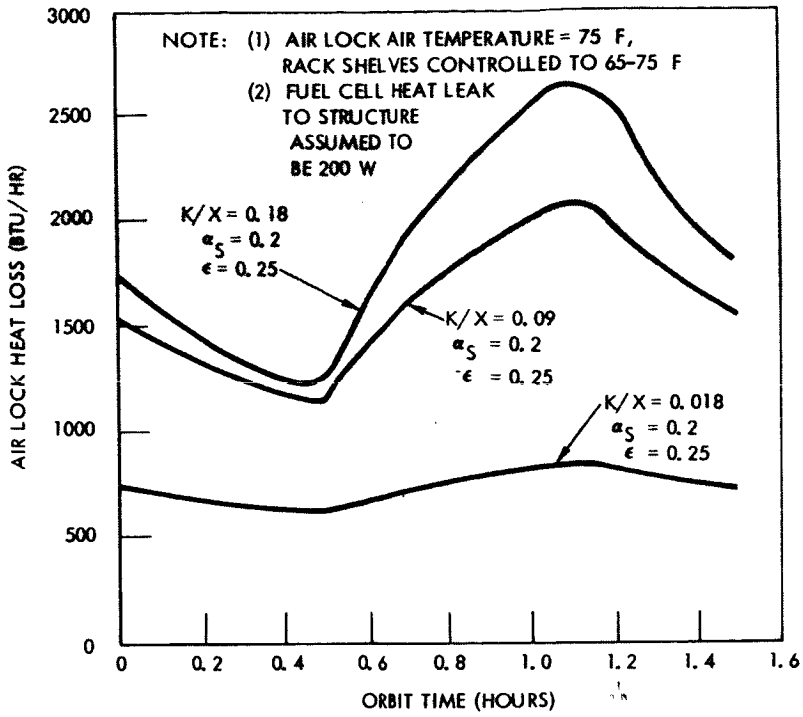


Figure 80. Air Lock Heat Loss for 200-Nautical-Mile, Earth Orbit, Solar-Oriented

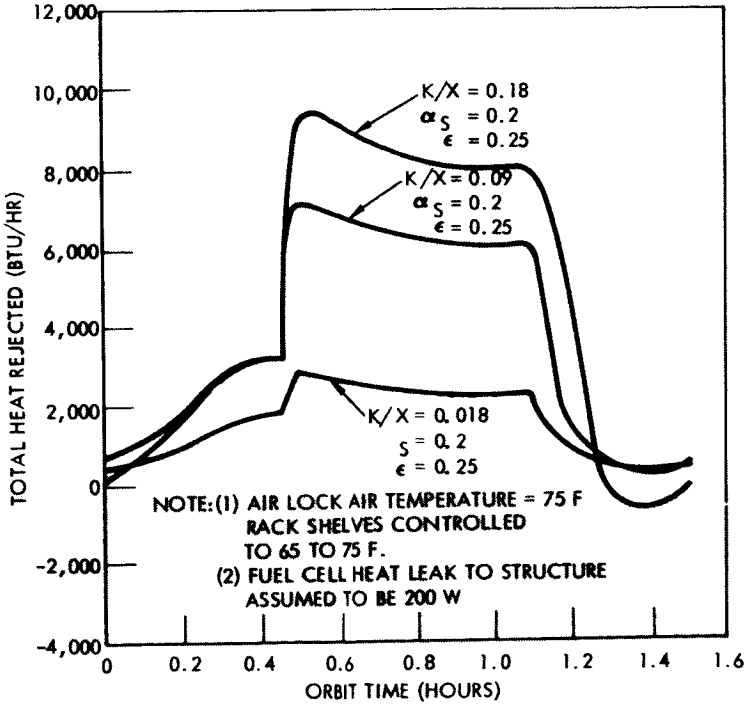


Figure 81. Rack Heat Loss for 200-Nautical-Mile Earth Orbit, Solar-Oriented

Table 38. Synchronous Orbit, Solar-Oriented

Case	k/x	Orientation	Air Lock Control (°F)	Shelf Control (°F)	Fuel Cells	Air Lock Heat Loss (Btu/hr)	Total Heat Loss (Btu/hr)
1	0.18	Toward sun	75	65 to 75	yes	1975	3954
2	0.09	Toward sun	75	65 to 75	yes	1783	3668
3	0.018	Toward sun	75	65 to 75	yes	1257	1831
4	0.18	Away from sun	75	65 to 75	yes	4167	9876
5	0.09	Away from sun	75	65 to 75	yes	3299	7760
6	0.018	Away from sun	75	65 to 75	yes	1747	3141
7	0.18	Away from sun	75	None	No	7205	7205

heat generated by personnel and experiments can be utilized to provide heating where needed, thus minimizing power requirements. It would also be advantageous to utilize waste heat from the fuel cells which would normally be dissipated by the EPS temperature control system. Insulation strength and location will depend to a great extent upon the particular mission at hand, particularly with regard to the type of experiments to be installed.

COMMAND MODULE THERMAL ANALYSIS

The purpose of the command module thermal analysis was to determine the amount of heat lost through the command module walls and establish the temperature gradients across these walls in the vicinity of the pressurized cabin for a number of Earth orbits and vehicle orientations. The amount of heat lost through the command module walls is necessary to conduct a heat balance on the command module to determine radiator requirements. The temperature at the command module inner wall was required because of the possibility of vapor condensation at temperatures below 50 F. The temperature at the ablator/heat shield interface was required because of the possibility of cracks in the ablator material at temperatures below -150 F.

The command module, divided into eight sectors, is shown in Figure 82. The cross-section through each sector also is shown in the figure. Since the ablator material thickness varies over the command module surface, average values were used as shown in Figure 82. The command module structure was thermally represented by a two-dimensional, analog network — a portion of which is shown in Figure 83. The material properties used to determine the conductors and capacitors shown on this network are presented in Table 39. It was assumed that these properties were independent of temperature. In addition to these assumptions, the following set of parameters were used for all cases:

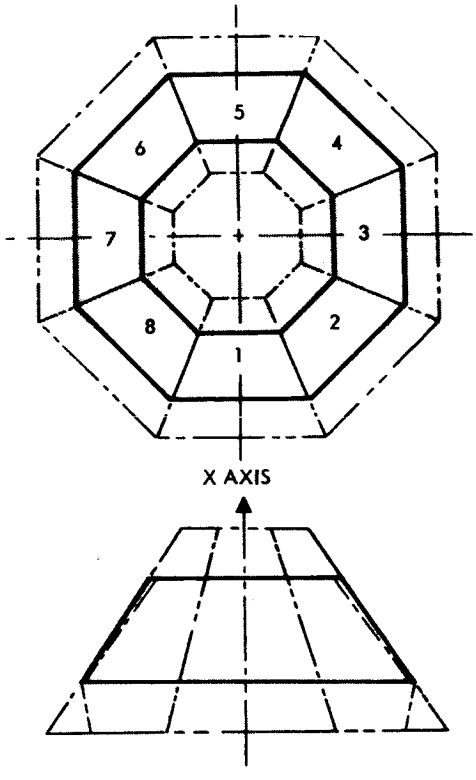
1. The temperature of the cabin oxygen was held to 70 F, the film coefficient between the oxygen and the cabin wall was assumed to be 1.0 Btu/hr-ft^2- F.

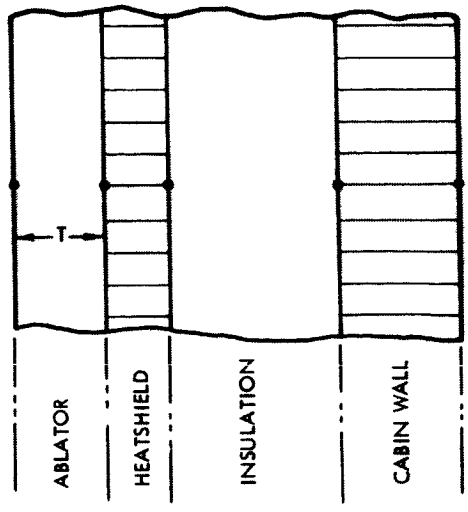
2. The solar absorbtivity (α_s) of the outer surface was assumed to be 0.20, the emissivity was assumed to be 0.90.

3. The shielding effect of the rack was neglected. The complexity of this problem precluded any effort to its solution in the limited amount of time available for the study.

A transient analysis was conducted utilizing a general heat transfer program designed for use on the IBM 7090/7094 digital computer. The heat fluxes imposed on the surface nodes were obtained from an orbital heating program which accounts for the direct solar, reflected solar and earth emission heat loads. The vehicle was allowed to orbit the Earth until the temperature and heat loss variations per orbit were nearly stabilized. Fifteen cases were computed and are tabulated in Table 40 which lists the orbit parameters (altitude, inclination, epoch, etc.). For each case, sector 1 was oriented such that it received a minimum of the incident heat flux. All cases computed were subjected to a transient analysis except case 15, which was computed for steady state conditions due to the relatively constant heat fluxes on all surfaces.

The average heat loss rates (and their related orbits) for each case are presented in Table 40. The variation of heat loss with orbit period for all cases (except 15) are presented in Figures 83, 84, and 85. The temperature drop across surfaces 1, 3, and 5 for typical orbits are presented in Figures 86 through 92. The time shown in these figures is the time in the orbit period at which the temperature of surface 1 is at or near the minimum value for the orbit.





SURFACE	T
1	0.75
2	0.82
3	0.91
4	1.13
5	1.50
6	1.26
7	0.92
8	0.82

Figure 82. Command Module Thermal Analog

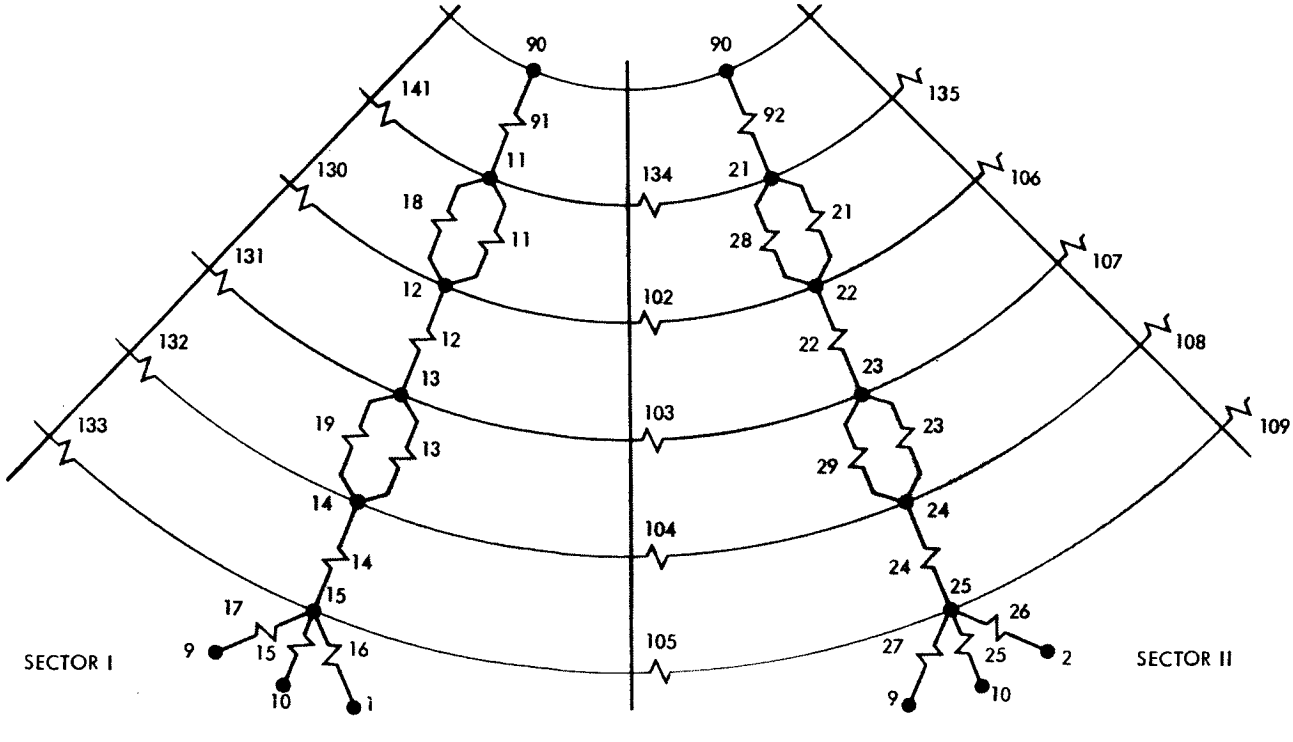


Figure 83. Typical Pattern for Sectors I Through VIII

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Table 39. Material Properties of Various Conductors and Capacitors

Material	Conductivity (Btu/hr-ft-F)	Heat Capacity (Btu/lb-F)	Density (lb/cu ft)	Emissivity
Ablator (Avcoat)	0.07	0.36	35.0	
Heat Shield (stain steel honeycomb)	8.8	0.12	5.4	0.7
Insulation (microquartz)	0.025	0.20	6.3	
Cabin Wall (aluminum honeycomb)	80.0	0.20	4.4	0.5

Table 40. Computer Run Matrix

Case No.	Altitude (n. mi.)	Inclination	Epoch	Right Ascension	Eccentricity	Orientation	Average Heat Loss (B/hr)	Orbit No.
2	200	28.5	80.0	0	0	Earth, Y axis ⊕	1232	9
3	200	28.5	80.0	0	0	Inert., X axis ♈	923	8
6	200	50.0	80.0	0	0	Earth, Y axis ⊕	1185	8
9	200	50.0	80.0	0	0	Inert., X axis ♈	935	8
19	200	50.0	80.0	0	0	Solar, Y axis ⊙	1193	9
11	200	81.5	80.0	0	0	Earth, Y axis ⊕	1161	7
10	200	81.5	80.0	0	0	Inert., X axis ♈	918	7
4	200	90.0	314	75.0	0	Earth, Y axis ⊕	1200	8
5	200	90.0	314	75.0	0	Inert., X axis ♈	1380	8
13	200	81R	284	0	0	Earth, Y axis ⊕	1151	7
12A	200	81R	284	0	0	Inert., X axis ♈	936	7
14	19,357	0	222	0	0	Earth, Y axis ⊕	1573	1
15	19,357	0	222	0	0	Solar, Y axis ⊙	1612	Steady state
16	700/200	50.0	80	0	0.065	Earth, Y axis ⊕	1212	6
17	700/200	50.0	80	0	0.065	Solar, Y axis ⊙	1242	7

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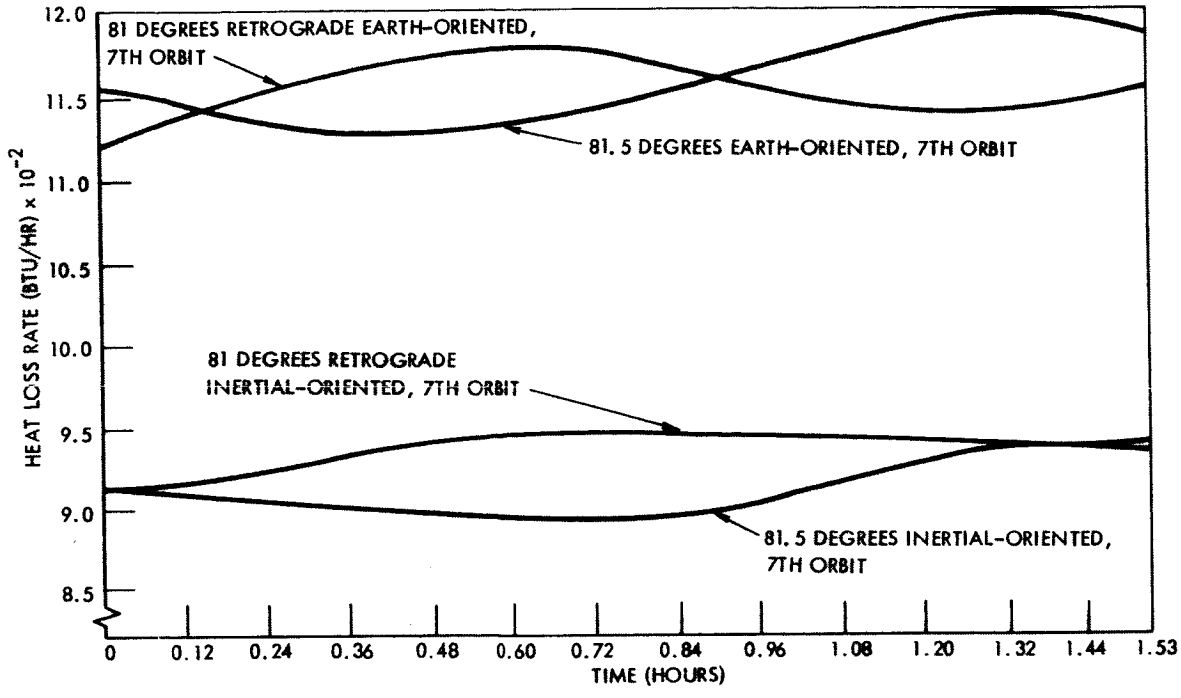


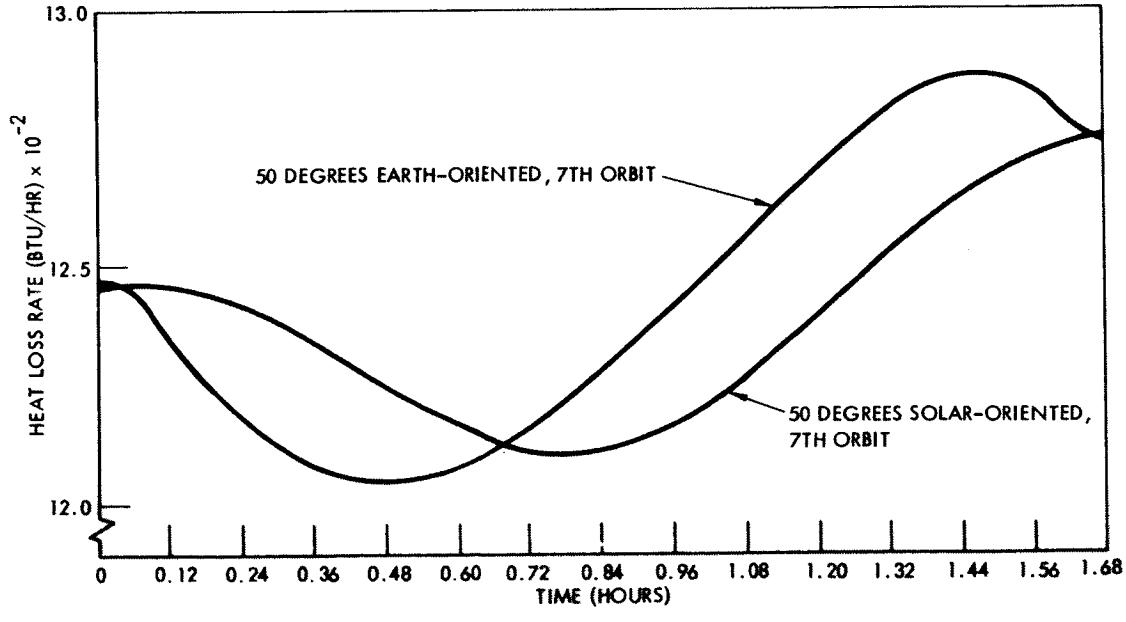
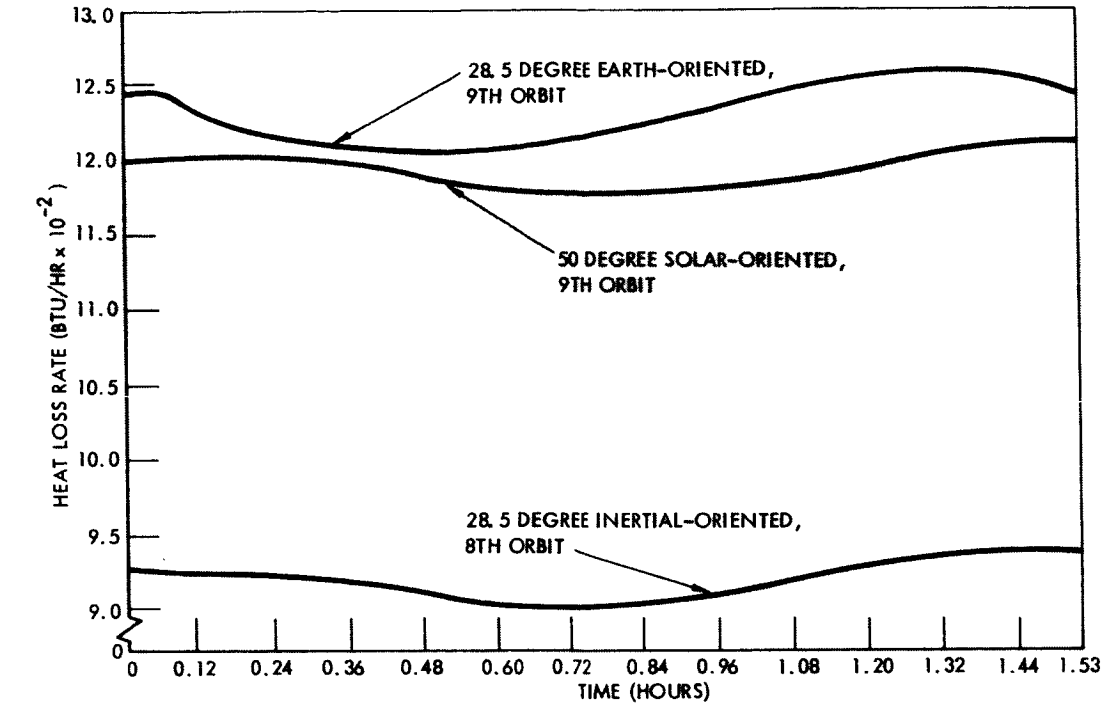


Figure 84. Transient Heat Loss per Orbit, 200-Nautical-Mile Orbit (Sheet 1 of 2)



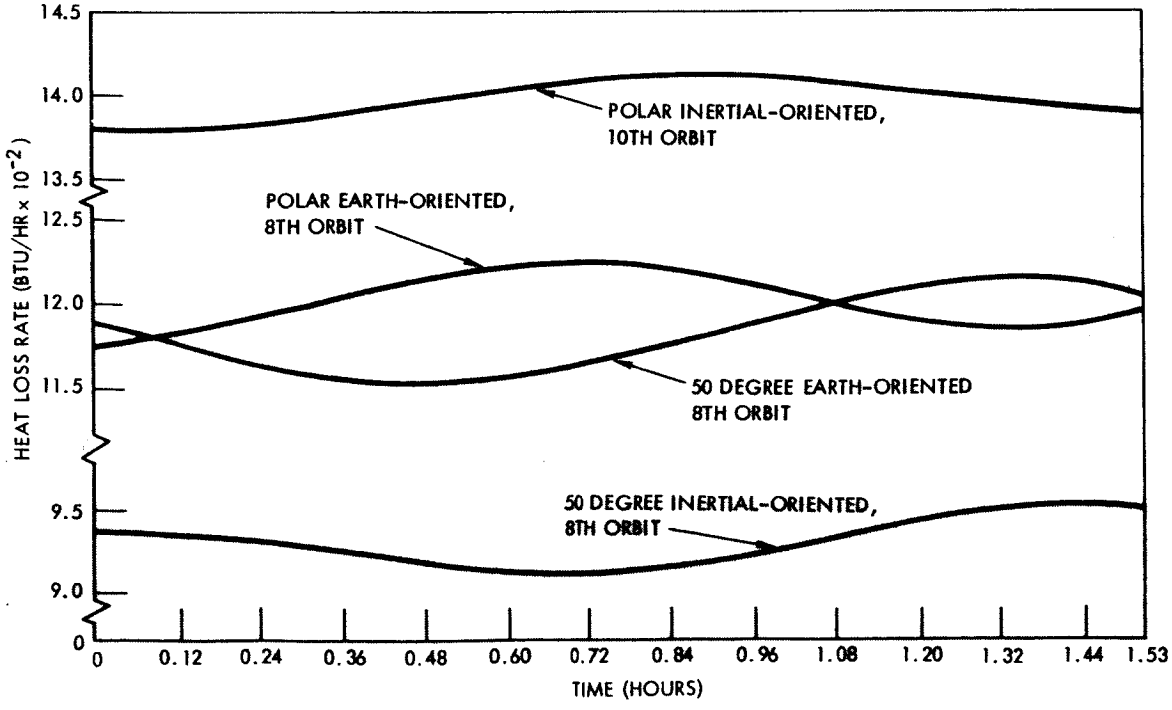


Figure 84. Transient Heat Loss per Orbit, 200-Nautical-Mile Orbit (Sheet 2 of 2)

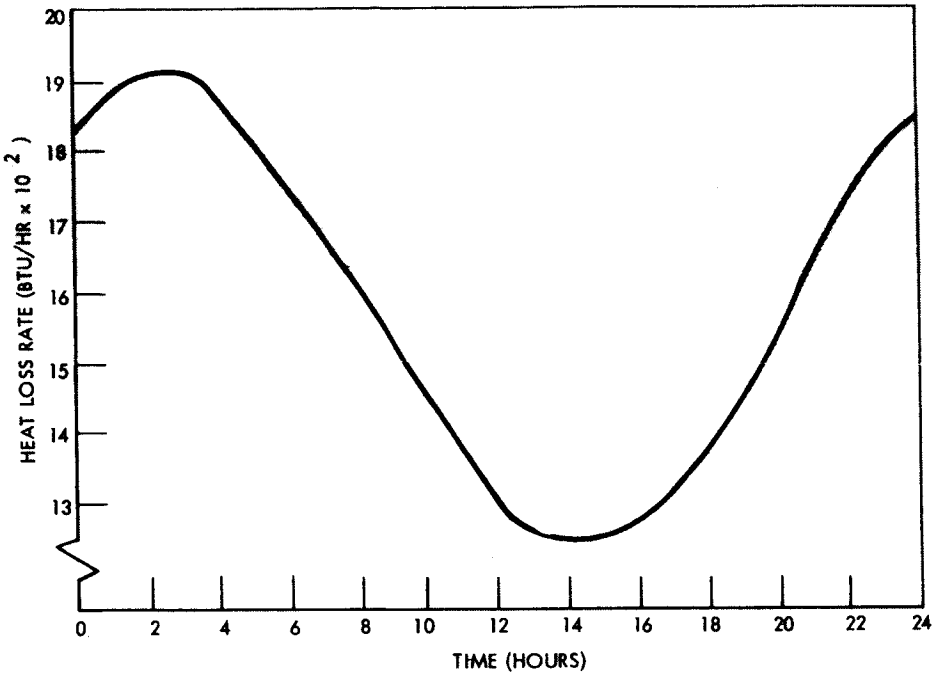


Figure 85. Heat Loss Versus Time (Synchronous) Orbit, Earth-Oriented ($t_{Initial}$ = 16.5 Hours)

For Case 15, the heat loss shown in Table 40 represents a heat loss that occurs when the temperature of the inner wall at sectors 1, 2, 3, 7 and 8 were held to 50 F. This was done because the results of the initial computer was indicated that the temperature of these surfaces fell below 50 F.

The surface temperature history of various surfaces are shown in Figures 93, 94, and 95 for Runs 2, 3, and 4.

For Runs 2, 3, 6, 9, 10 and 11, an inspection of the average heat loss rates (Table 40) indicate that inertially oriented vehicles lose approximately 200 Btu/hr less heat than the Earth-oriented vehicles. The reason for this is that the vehicle x axis is essentially sun oriented; therefore, all surfaces are exposed to direct solar radiation when not in the Earth's shadow. Runs 2, 6, and 11 show that the effect of inclination angle is small; as the angle increases, the heat loss rates decrease somewhat, but not appreciably.

The maximum heat loss rate (1915 Btu/hr) occurred during Run 14, the synchronous, Earth-oriented case. The minimum heat loss rate (895 Btu/hr) occurred during run 10 -- a high inclination, inertially oriented case.

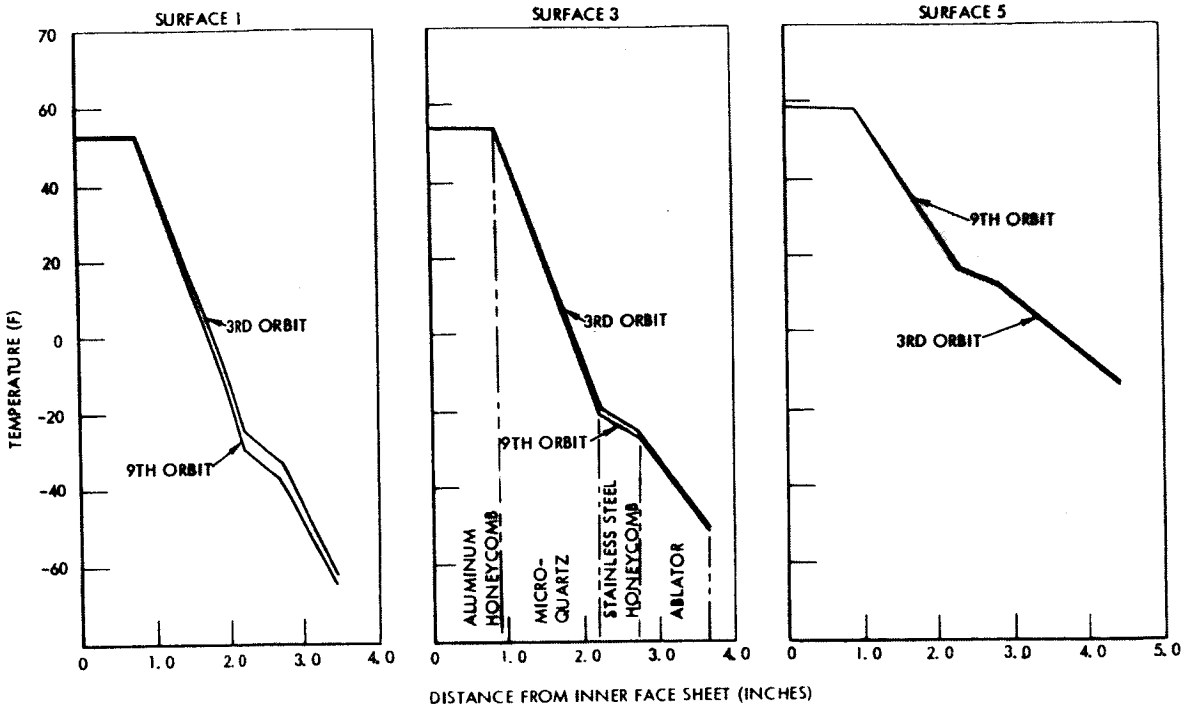


Figure 86. Temperature Drop Across Command Module Structure, 200-Nautical-Miles, 28.5 Degrees, Earth-Oriented

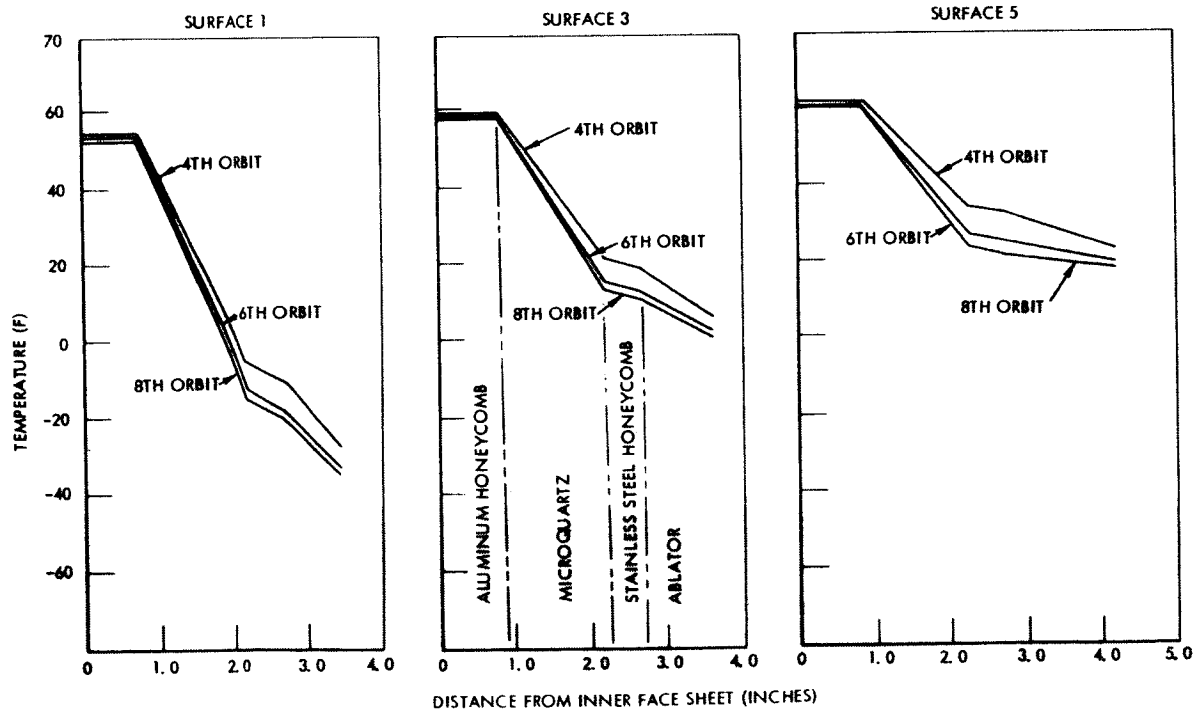


Figure 87. Temperature Drop Across Command Module Structure 200 Nautical Miles, 81.5 Degrees, Inertial-Oriented

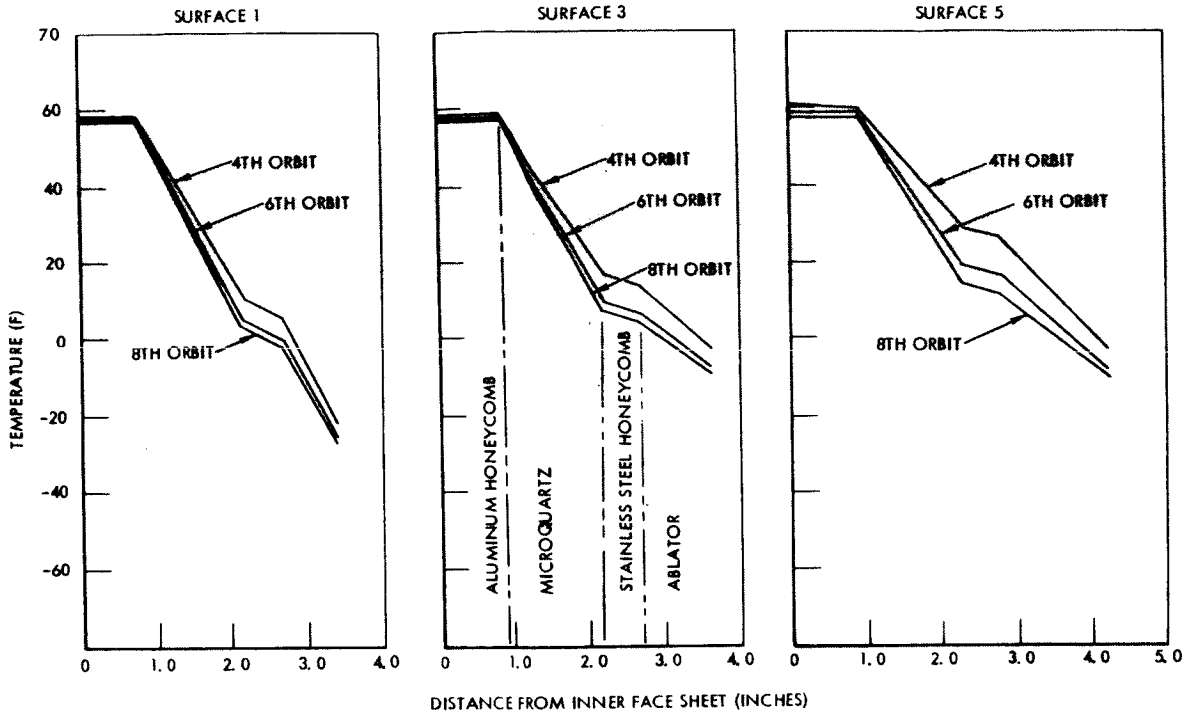


Figure 88. Temperature Drop Across Command Module Structure, 200-Nautical-Miles, 81 Degrees Retrograde, Inertial-Oriented

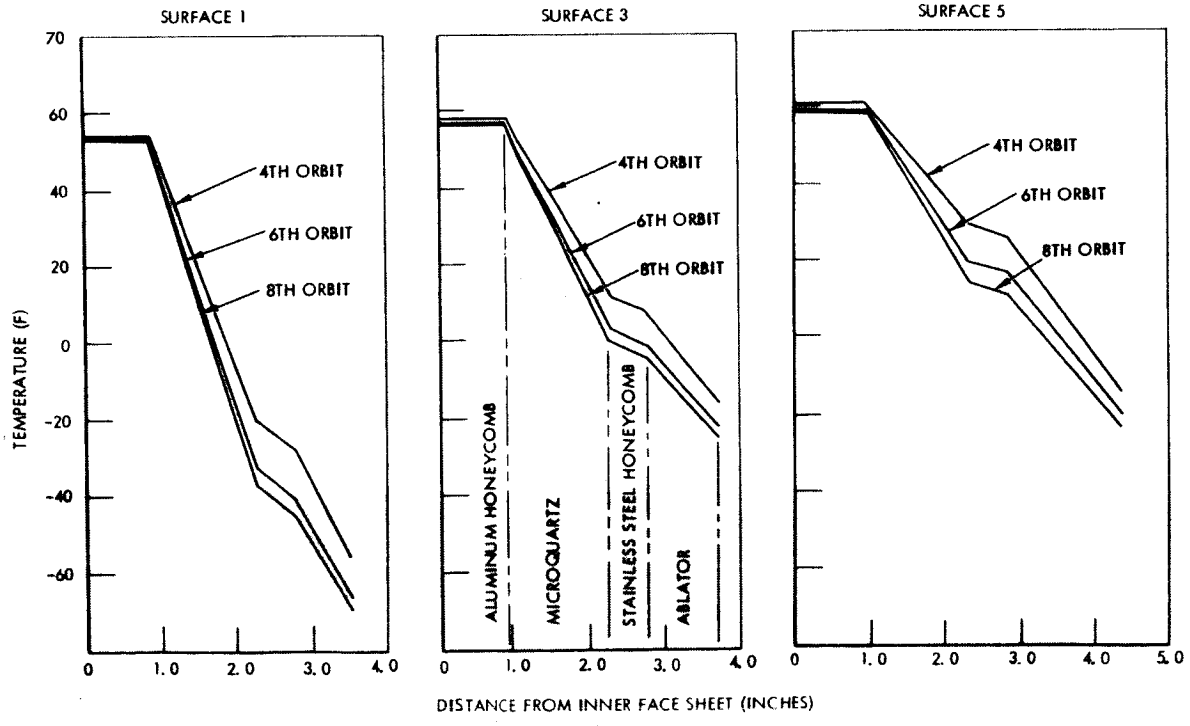


Figure 89. Temperature Drop Across Command Module Structure, 200 Nautical Miles, 81 Degrees Retrograde, Earth-Oriented

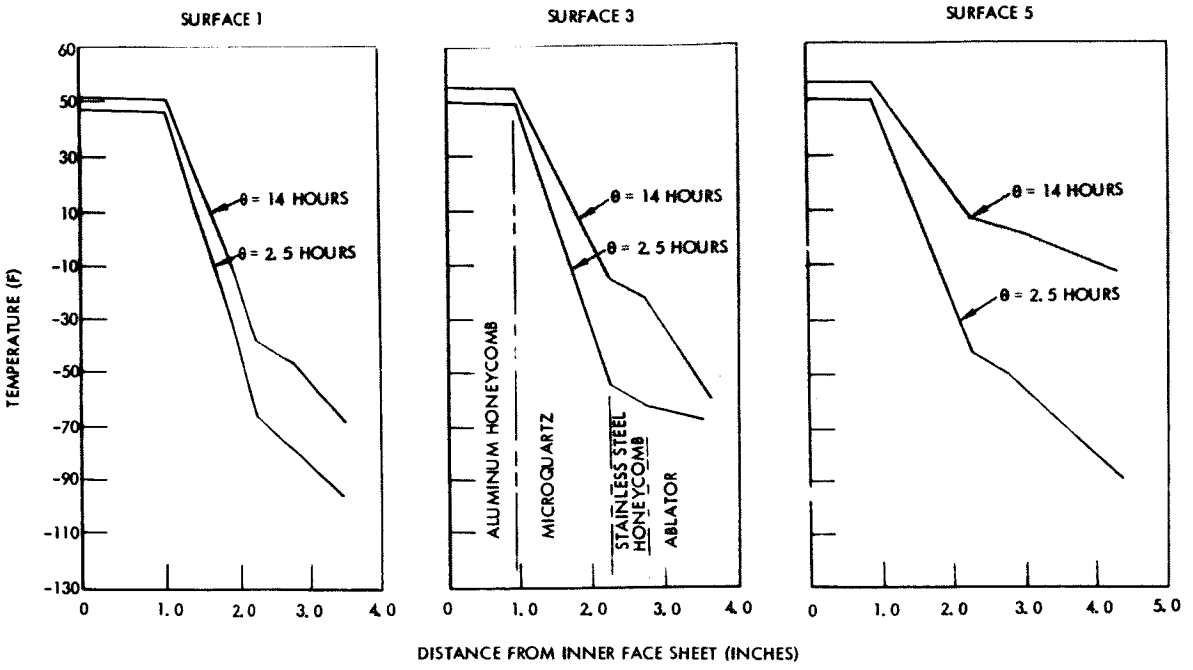


Figure 90. Temperature Drop Across Command Module Structure, Synchronous Orbit, Earth-Oriented

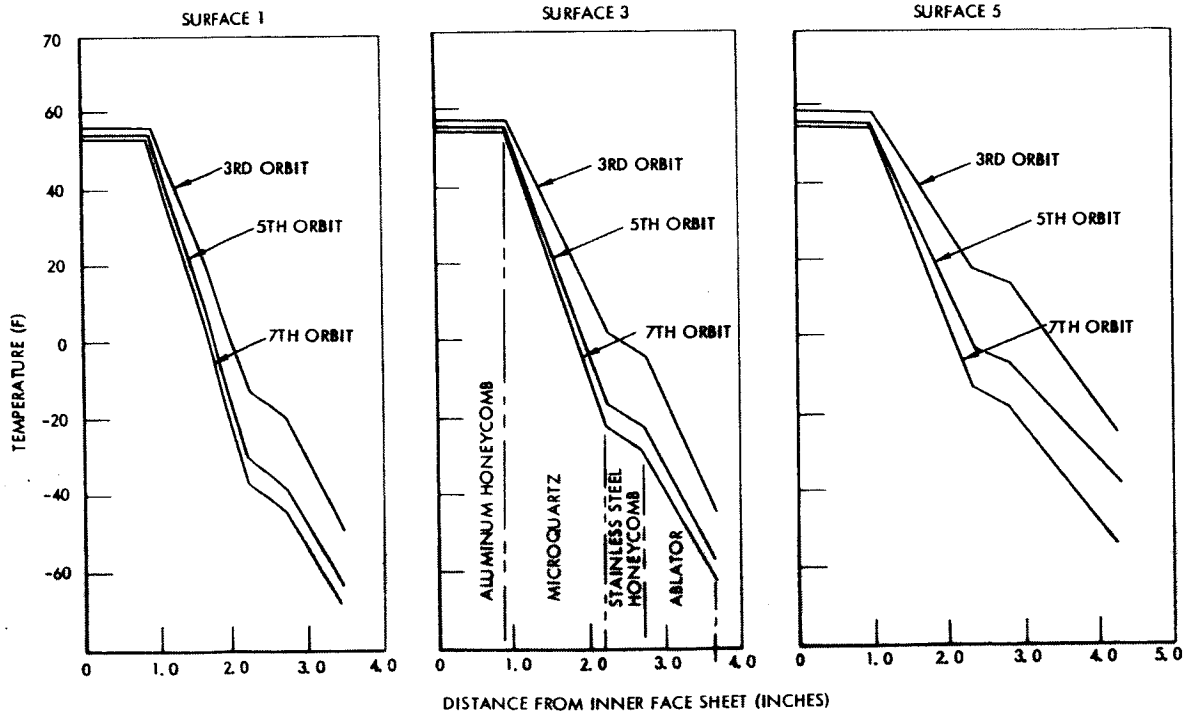


Figure 91. Temperature Drop Across Command Module Structure, Run 16, 700/200 Nautical Miles, 50 Degrees, Solar-Oriented

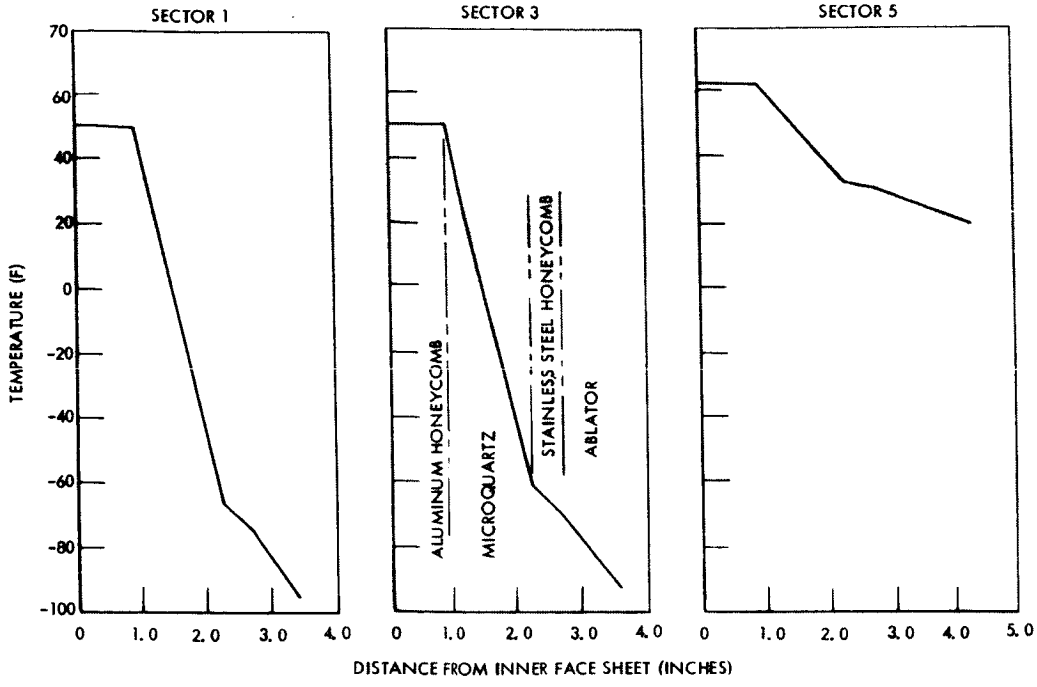


Figure 92. Temperature Drop Across Command Module Structure, Steady-State, Synchronous-Orbit, Solar-Oriented

An inspection of the temperatures at the ablator/heat shield interface indicates that there is no problem. The lowest interface temperature encountered was -77 F which occurred during run 14.

The possibility of condensation occurring at the inner wall may be a problem, especially for the synchronous orbit case (Run 14). During this run, the temperature at the inner wall is approximately 48 F. (See Figure 90). In fact, the inner wall temperature at sector 1 approaches 50 F for most cases. This situation may be further aggravated when the shielding effect of the rack is accounted for in future studies. The problem may be alleviated by orientating the vehicle to reduce heat losses (i.e., sun-oriented), but since this places a constraint on the mission this may not be an acceptable solution to the problem.

To define and help solve the problem, shielding effects, additional heat loads by men and equipment, components mounted to the cabin walls, etc., should be studied in greater depth. This type of analysis is considered beyond the scope of this study.

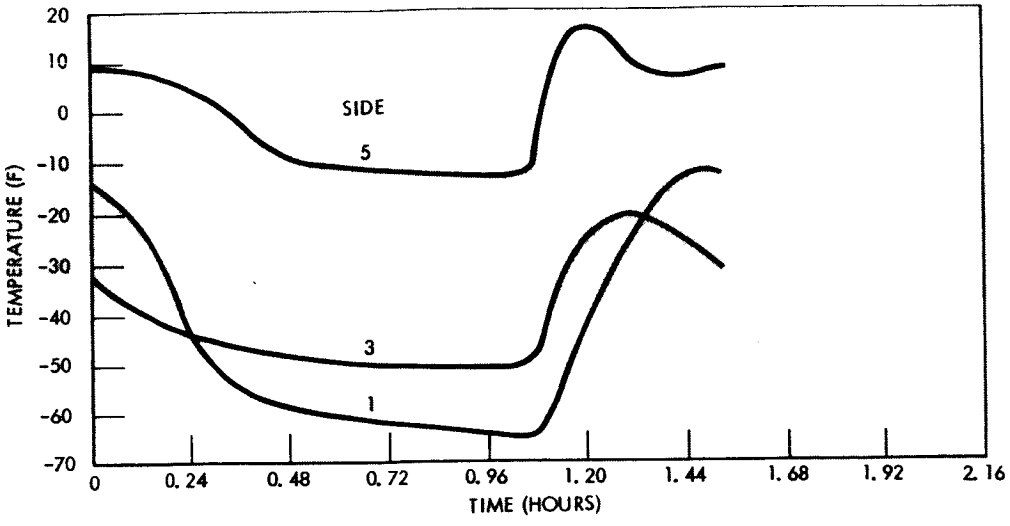


Figure 93. Surface Temperature Variation, 9th Orbit, Run 2

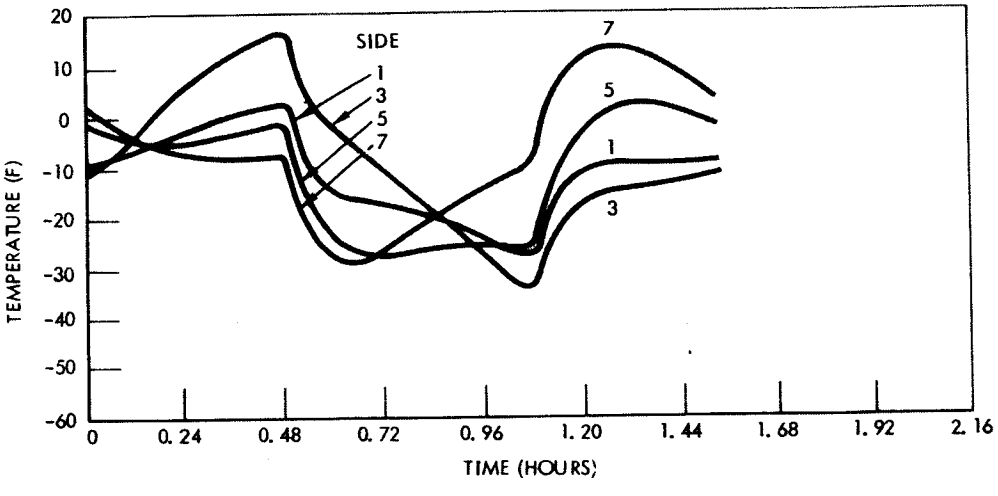


Figure 94. Surface Temperature Variation, 8th Orbit, Run 3

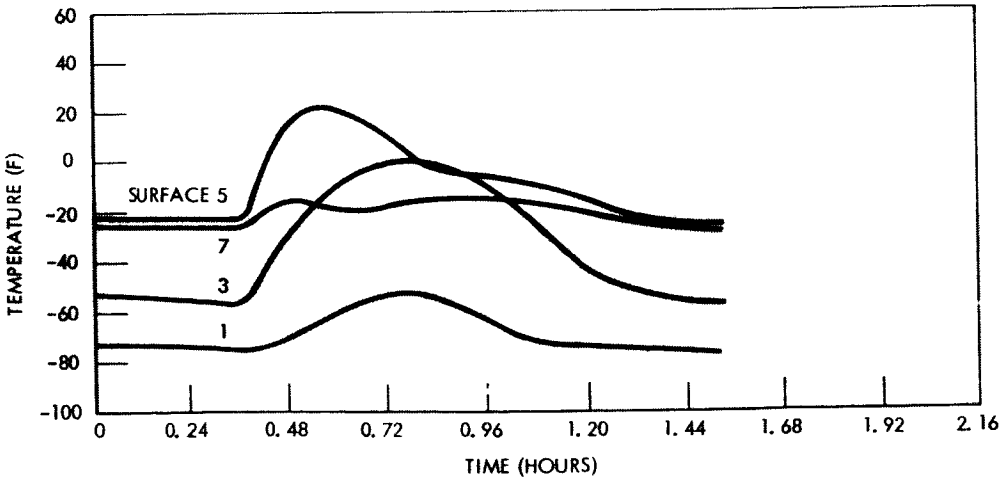


Figure 95. Surface Temperature Variation, 8th Orbit, Run 4

PARACHUTE COMPARTMENT THERMAL ANALYSIS

Nylon parachute material is presently qualified to maintain structural integrity at -65 F. The purpose of this analysis was to determine if this temperature limit was exceeded and, if so, the amount of heat necessary to maintain temperatures above -65 F. In addition, it was necessary to determine the temperature of the ablator/heat shield interface due to the possibility of cracks developing in the ablator material at temperatures below -150 F.

The parachutes are stowed in three of the four compartments adjacent to the command module tunnel. Because of the presence of the rack, the outer surface of the parachute compartment may be shielded from incident thermal radiation a significant percentage of the time -- depending upon vehicle orientation. It was assumed, therefore, that the "worst case" conditions existed during these periods and that the temperatures would reach steady state conditions.

A cross-section of the compartment is shown in Figure 96. It should be noted that insulation has been placed over the parachute. It was assumed that the insulation would be required because recent Apollo studies indicated the need for thermal protection caused by the heating effects of reentry.

The thermal network used is shown in Figure 97. The material properties used to determine the value of the conductors shown in this model are listed in Table 41.

Initially, computer runs were made which reflected the use of an outer surface form factor to space of 1.0 and an emissivity of 0.9. The results of these computations indicated that the form factor assumption was too conservative because skin temperatures exceeded -300 F. Subsequent computer runs were made using a form factor of 0.4, which accounts for the effect of the rack shielding the surface from deep space. In addition, the emissivity of the outer surface was revised to 0.25 to reduce heat losses.

The temperature of the oxygen in the cabin and the tunnel was held to 70 F. The film coefficient between the air and the cabin bulkhead and (initially) the tunnel wall was assumed to be 0.5 Btu/hr=ft^2-°F. During subsequent computer runs, the film coefficient between the oxygen and the tunnel was set to zero to reflect the condition that exists when the tunnel is void of oxygen.

A number of computer runs were made which indicate the effects of varying the emissivity of the surface of the insulation over the parachute and the surface of the insulation under the heat shield. A value of 0.7 was used for both surfaces when uncovered; 0.06 when covered with "shiny"

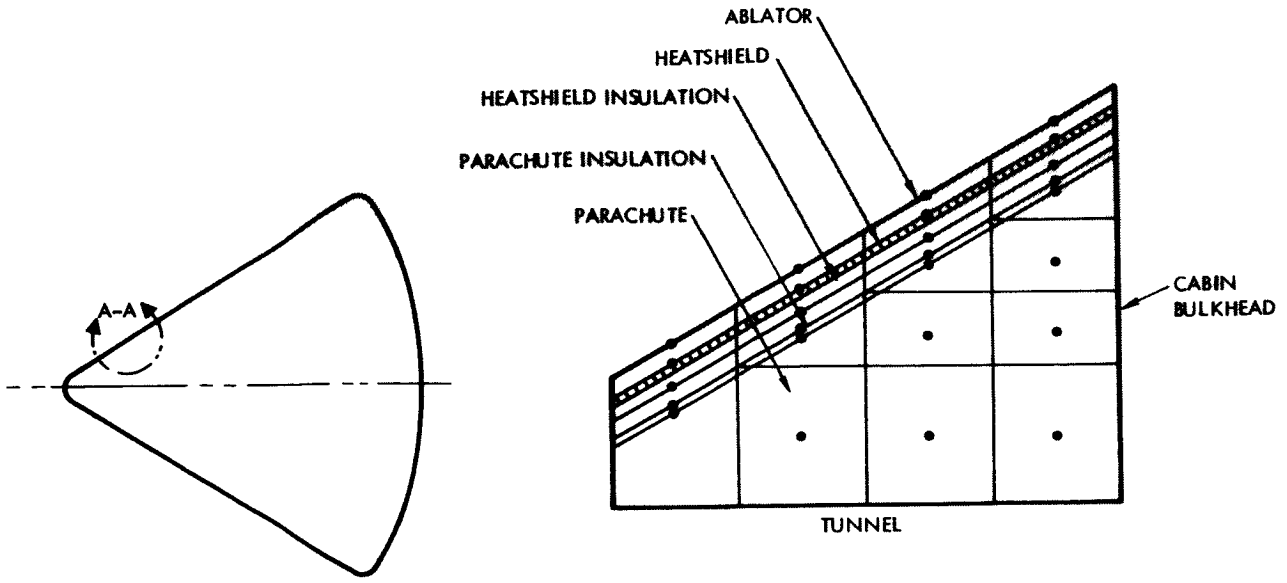


Figure 96. Parachute Compartment Structural Schematic

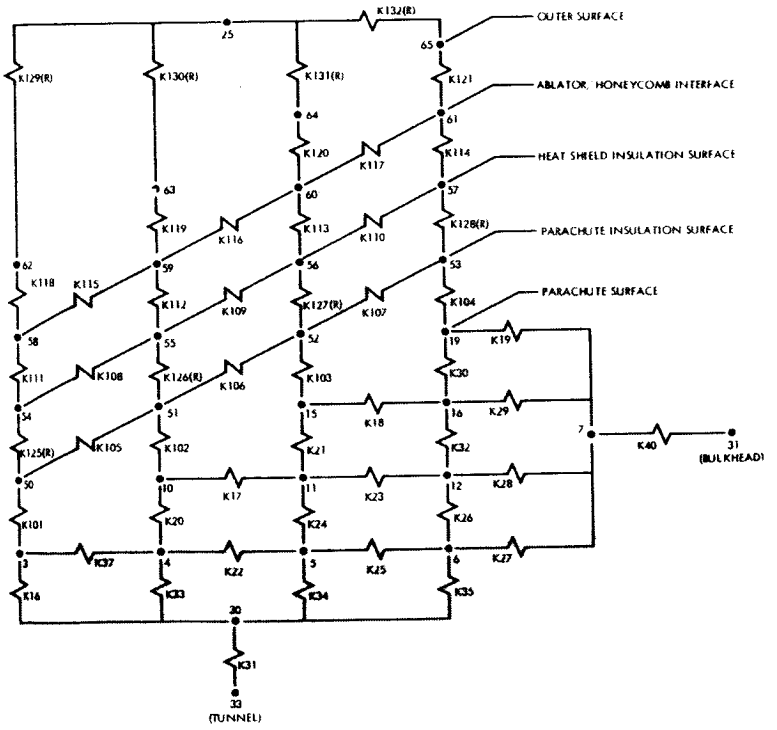


Figure 97. Thermal Network, Parachute Compartment

aluminum, and 0.9 for a "black" radiator that may be used to maintain temperature control of the surfaces under consideration.

Table 41. Material Properties

Material	Conductivity (Btu/hr-ft-°F)
Ablator (Avcoat)	0.04-0.07
Heat shield (stainless steel)	8.8
Insulation (Microquartz)	0.025
Insulation (H. I. Thompson)	0.002
Parachute (nylon)	0.08

Initially, a value of 0.07 Btu/hr-ft-ft-°F was used for the conductivity of the ablator material. However, at the temperatures resulting from the initial computer runs, the conductivity is approximately 0.04 Btu/hr-ft-°F. This value was used for subsequent computations.

The pertinent results of a number of steady-state computer runs are shown in Table 42. The effects of changes in the insulation surface radiation properties are reflected in the first three computer runs. Initially, for an emissivity of 0.7 for both surfaces, the temperatures of the ablator/heat shield interface are just within the temperature limit of -150F; however, the surface temperature of the parachute is below the minimum allowable temperature of -65 F. When the emissivity values are decreased to 0.06 for both surfaces (Run 3), the parachute temperatures fall within the acceptable limits but the ablator/heat shield interface temperature limit is exceeded.

The same effects are noted in runs 4, 5, and 6 which are indicative of the "void" condition of the tunnel. That is, an increase in emissivities results in an increase in energy transport which, in turn, increases the interface temperature and decreases the parachute surface temperature. However, at no time are the material temperatures within the acceptable limits.

[redacted]

Case 7 shows the effects of the decreased (to 0.04) conductivity of the ablator material. Although the emissivity of the parachute insulation surface is increased slightly (to 0.9), the resulting temperature change is due primarily to the change in conductivity. The increased emissivity value is indicative of a condition that exists when a "black" heat source (i.e., a resistance heater) is added to the parachute insulation surface.

Cases 8 through 11 show the effects of a step increase in heat input to the parachute insulation surface. It can be seen that a total of approximately 20 Btu/hr is required to maintain the temperatures within their acceptable limits.

The results of the study indicate that for the three parachute compartments, approximately 60 Btu/hr (18 watts) is required to maintain temperature control of the material under the worst-case conditions. This rate will be intermittent, depending upon external incident heat fluxes encountered during orbit. However, for the best case conditions, the temperatures are within the specified limit of -65 F with no thermal control.

Table 42. Computer Run Matrix

Run No.	q (Btu/hr)	h Tunnel Btu/hr-ft^2-F	ϵ Parachute Insulation	ϵ Heat Shield Insulation	k Ablator Btu/hr-ft-F	Parachute Insulation Surface Temp, °F				Heat Shield Insulation Surface Temp, °F			
						T_{19}	T_{15}	T_{10}	T_3	T_{61}	T_{60}	T_{59}	T_{58}
1	0	0.5	0.7	0.7	0.07	- 88	-116	-115	- 87	-140	-145	-146	-141
2	0	0.5	0.7	0.06	0.07	- 46	- 82	- 80	- 45	-165	-168	-169	-166
3	0	0.5	0.06	0.06	0.07	- 16	- 51	- 50	- 15	-185	-187	-187	-185
4	0	0	0.06	0.06	0.07	- 26	- 84	-103	- 91	-201	-205	-208	-210
5	0	0	0.7	0.06	0.07	- 58	-117	-136	-124	-186	-192	-197	-199
6	0	0	0.7	0.7	0.07	-109	-155	-169	-163	-167	-177	-185	-187
7	0	0	0.9	0.7	0.04	-106	-150	-164	-158	-160	-170	-178	-180
8	5.0	0	0.9	0.7	0.04	- 83	-118	-130	-125	-134	-143	-149	-151
9	10.0	0	0.9	0.7	0.04	- 63	- 92	-102	- 98	-114	-121	-127	-128
10	15.0	0	0.9	0.7	0.04	- 43	- 68	- 76	- 73	- 95	-101	-105	-106
11	20.0	0	0.9	0.7	0.04	- 25	- 46	- 52	- 50	- 77	- 82	- 86	- 87

[redacted]

[redacted]

ENVIRONMENTAL CONTROL SYSTEM

The environmental control system (ECS) -- consisting of two principal subsystems, the atmosphere control subsystem and the thermal control subsystem -- provides and maintains a suitable environment for the crew and certain equipment. Suitable environment for equipment becomes one of the important objectives of this study inasmuch as the proper operation of experiments is essential to achieving mission objectives.

The study, an addendum to the Extended Apollo Systems Utilization Study, is based on the Block II Apollo vehicle and its variations. Therefore, the present ECS system concept is that which is utilized in the Block II Apollo. The selection of the subsystems for the Apollo ECS has been fully justified by design studies of Project Apollo; the applicability of the system for the present missions has been amply demonstrated in previous studies, reported in SID 64-1860. Therefore, the present study was mainly devoted to the integration of the ECS requirements of the external device (rack) with the command module systems. Discussion of the requirements of the four configurations follows.

CONFIGURATION 1

Configuration 1 is based on the use of the Block II CSM with no subsystem changes -- with the exception of those necessary to effect the interface with the external device. All mission durations are thus limited to 14 days or less. The external device (rack) features two areas of concern to the ECS system: an airlock with a volume of 216 cubic feet; and an unpressurized area for locating experiments, expendables, and some subsystems. The air lock will be considered here. The temperature control of the unpressurized area is discussed earlier. The requirements for the airlock in Configuration 1 are as follows:

1. An assumed oxygen atmosphere at a total pressure of 5 psia.

2. Metabolic oxygen will be supplied by the CSM, except when the connecting hatch between the command module and the air lock is closed.

3. Oxygen for repressurization will be stored at high pressure in a tank in the unpressurized area of the rack. This tank could also serve as the source of metabolic oxygen when the command module rack connecting hatch is closed.

4. Provisions are made for both temperature and pressure control of the air lock atmosphere, separate from the CM controls.

5. There will be a retractable duct between the CM and the airlock for intermodule atmosphere circulation, for CO_2 removal and humidity control of the airlock

6. Provide for contaminant control of the airlock atmosphere.

Figure 98 shows schematically how these requirements can be met.

At launch, the airlock is purged and pressurized with pure oxygen at atmospheric pressure. As the vehicle gains altitude after launch, the airlock pressure bleeds down through the pressure control valve. This valve shuts off when the internal pressure reaches 5 psia. After docking, the crew can open the interconnecting hatch, install the retractable duct, and turn on the intermodule circulation fan. Circulation of the atmosphere insures provision of metabolic oxygen from the normal CSM source. The circulation rate is set by the humidity control requirement. This requirement is the limiting factor since it requires a greater rate than for CO_2 control. Calculations show a flow rate of 61 CFM is necessary, requiring a 42-watt fan.

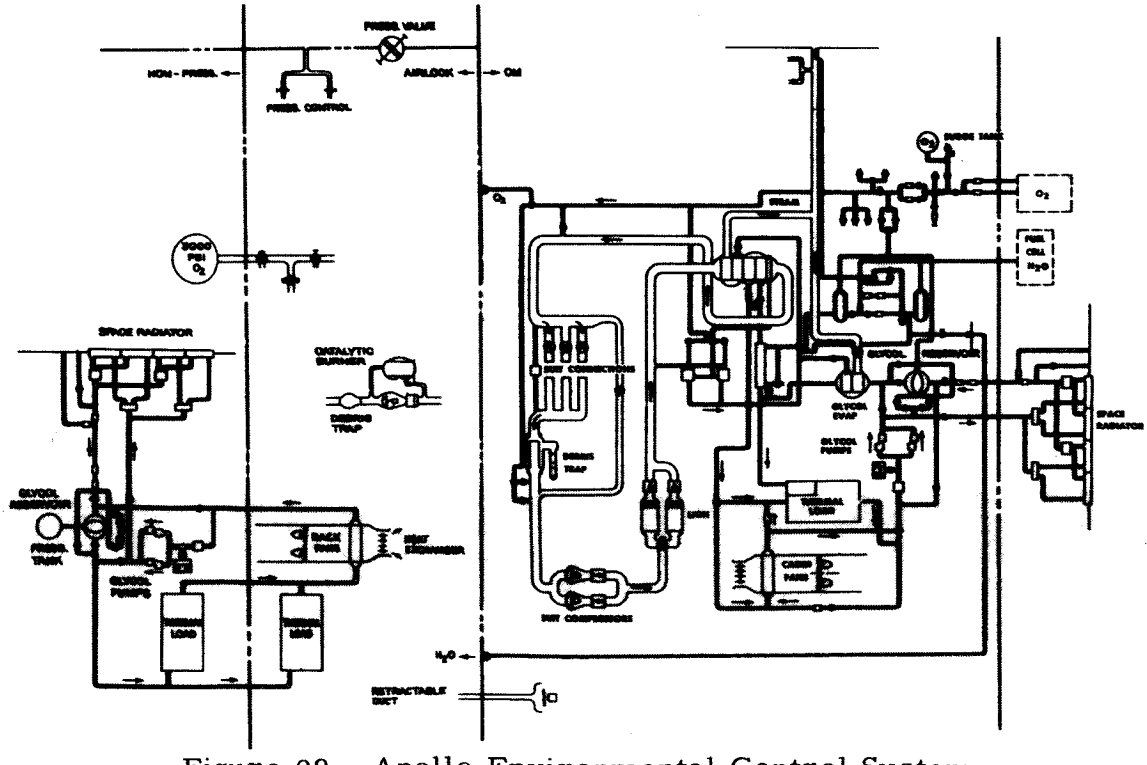


Figure 98. Apollo Environmental Control System Configuration 1

Extravehicular operations necessitate repressurization of the airlock. The crew member dons his pressure suit which is equipped with a portable life support system. He then closes off the air lock from the command module, manual override on the pressure control valve, and depressures the airlock by opening the air lock pressurization valve. Oxygen for repressurization is stored at 3000 psia in a tank mounted outside the airlock. A line from this tank carries oxygen into the air lock through a pressure-reducing regulator and a manually operated valve which repressurizes the air lock. For operation with the interconnecting hatch to the command module closed, metabolic oxygen also comes from this line through a pressure regulator set at 5 psia. This regulator is located downstream of the pressure reducer. The oxygen tank carries 84.5 pounds of oxygen gas at 3000 psia, sufficient for 15 repressurizations.

Air lock atmosphere temperature control is accomplished by means of a fan-eat exchanger combination similar to the command module fan equipment. Preliminary thermal analyses show that there is a minimum heat loss of 700 Btu per hour from the airlock with the vehicle in a 200-n mi Earth orbit. The magnitude of the heat loss depends on vehicle orbit, orientation, airlock wall structure, surface finish, and internal conditions. At the present time, the complete air lock thermal analysis is not available and worst conditions are not yet known. For the present, the fan design was based upon circulation rates which are considered proper for comfortable ventilation of the crew, without considering the flow rate through the heat exchanger. Using a flow rate of 60 cfm, and a characterization factors $\Delta P/\sigma$ of 3, the required fan power rating is 12 watts. For reliability, two fans should be used.

Air lock pressure control is maintained by the command module with the interconnecting hatch open; with the hatch closed, the high pressure supply takes over. Excess pressure is relieved by the pressure control valve.

Contamination control within the air lock is provided by a debris trap, a catalytic burner, and a chemisorber bed. The schematic of Figure 98 shows a separate fan which circulates about 1 pound per hour through the catalytic burner assembly. Further study is required to establish whether the contamination system should be integrated with the cabin fan, or maintained as an independent item. The debris trap removes any liquid and solid particles from the atmosphere, and the high molecular weight contaminants are absorbed by activated charcoal contained in the lithium hydroxide canisters. All combustible contaminants are catalytically oxidized by the catalytic burner, while the chemisorbent bed removes those which are not absorbed by charcoal. It is estimated that the catalytic burner requires an input of 8 watts for its electric heater.

CONFIGURATION D'

This configuration is a Block II CSM plus a rack. The mission duration is 30 days; this additional 16 days over the Apollo mission will be achieved by placing extra expendables in the rack. The ECS schematic is the same as shown in Figure 98 with the following additions:

1. Three Block II cryogenic oxygen tanks are placed in the rack. Piping from these tanks to the rack air lock provides backpack refill capability, and air lock repressurization through a surge tank.

2. A hardline from the rack cryogenic tanks carries oxygen at 100 psia to the command module for metabolic use and total pressure control. This line connects into the command module oxygen system immediately downstream of the pressure reducer regulators. This regulator reduces the pressure of the service module oxygen from 950 to 100 psia.

3. Because of the increased mission duration, a third compressor must be included in the command module suit loop.

Control of pressure, temperature, and atmosphere purification for the airlock are the same as for Configuration 1. The 3000 psi repressurization tank is also included in Configuration D' to satisfy total repressurization cycle requirements.

CONFIGURATION C

Configuration C is the Apollo X CSM with the addition of a rack. It has an orbital life of 45 days. The ECS is similar to Apollo X except that the atmosphere is oxygen at 5 psia instead of the dual gas atmosphere. The schematic diagram of the system is therefore the same as shown in Figure 98 with the 3000 psia oxygen tank deleted. Control of the rack airlock atmosphere pressure and temperature is the same as Configuration 1. However, the airlock in Configuration C is repressurized from the CSM oxygen supply. This is accomplished through the 100 psi supply line connected to the command module oxygen system downstream of the pressure reducer.

Modifications are necessary for some equipment items because of the increase in mission duration — from the 14-day Apollo Block II to the 45-day AES-ECS. Suit compressors, cabin fans, and glycol pumps must be equipped with longer life bearings, and a third redundant compressor must be added to the command module suit loop.

To allow room for experiments, some of the expendables—such as lithium hydroxide, food, medical supplies and life support items—have been moved from the CM to the unpressurized area of the rack.

CONFIGURATION D

Configuration D is basically a Block II CSM and a rack, with additional expendables on the rack to increase the mission duration to 45 days. This is accomplished by using Apollo X cryogenic oxygen tanks in the rack; these tanks are initiated after the service module 14-day supply is expended. The difference between Configuration D and D', then, is that D uses Apollo X tanks and D' uses Block II tanks. The difference between Configurations D and C is as follows: D has Block II tanks in the service module and Apollo X tanks in the rack; C has only Apollo X tanks in the service module. All three configurations, (C, D, and D'), have the third suit loop compressor in the command module. Configuration D, however, having the larger Apollo X tanks in the rack, does not have the 3000 psi repressurization tank found in both D' and 1.

Because of the ground rule which stipulates that the atmosphere in all configurations must be oxygen only, there is actually no difference in concept between the Block II and the Apollo Extension Systems command module ECS systems. The AES system has rotating components modified for longer life, an extra suit compressor, and the recirculation blower with its retractable duct; however, it is schematically the same as the Block II. All ECS changes or modifications are necessary as a result of the presence of the rack and the longer life requirements.

RACK THERMAL LOOP

The rack has a partial ECS system installed within the air lock. This system is sufficient to circulate the atmosphere, control its temperature and pressure, control contaminants, and provide equipment thermal control. There is no pressure suit loop nor as yet any connections for suit umbilical hoses, although it might be advisable to include the latter.

ECS components in the rack and their functions have been discussed above. There is also a thermal control loop, shown in Figure 98 which consists of a closed liquid loop and a space radiator. Cold liquid from the radiator is used to cool the experiments mounted on shelves in the rack and to cool equipment located in the airlock. The warm liquid then passes through the air lock heat exchanger and transfers heat to the atmosphere, thus making up for the heat lost through the air lock walls. The liquid is then pumped to the space radiator for heat rejection. The quantitative treatment of the cooling system was given earlier in the thermal analysis section.

LIFE SUPPORT SYSTEMS

The deviations from the basic Block II configuration in the Life Support and Crew Systems areas are discussed in this section. Following a systems definition, the influence of the various missions upon the systems are discussed with relative observations of how these influences may affect the normal operations, both in systems operation and human performance. Where there seems to be considerable deviation from the applicable systems, these deviations are identified for follow-on study.

FOODS AND FEEDING

The foods handling system is comprised of a vehicle storage volume containing the coded meal packets, the preparation equipments, and a waste storage volume. Areas of concern are the diet definitions in caloric values and volume ratios, the physiology of food consumption, the relationship of caloric need and work performance, the management and treatment of the foods wastes residue and the management of the emptied containers.

The present 2500-2800 Kcal diet will suffice for all missions in which the crew duty is mainly that of systems operation and monitor. If the crewman is to be required to do extensive manual activity, or EVA, his metabolic requirement will rise at some ratio in a relation to the work. Present analysis of the work performance show that the lower caloric limit may be 3000 Kcal/man-day as may be influenced by the work performance and/or experiments. The variance of the foods system will ultimately be dependent upon the physiology of the participating crew compared to light to moderate work activity of the Apollo crew which now requires about 2500-2800 Kcal per man day. It is presently assumed that the man will be his own guide to his increased need of nutrition. The use of measuring systems or determining schemes for space use cannot now be determined. The accepted laboratory practice in determining the response to food metabolism is through the blood tests, gas analysis, etc., and must be studied for vehicle application before foods experiments in zero-g can be made totally valid. There are studies that will define the acceptability of on-board blood tests, but at this time, there is no assurance that this will, in fact, be a part of the present missions. For the most part, the results of body fluid analysis is a subject of post flight analysis. It is more probable that gas analysis systems can be more easily adapted.

Certain considerations are to be made for the foods management during the 45-day, 90-day, and 135-day extended occupancy missions (218, 219, and 523). These will involve the resupply contingencies, the emergency supply storage, the waste management and its post mission disposition (return logistics considerations), etc. As the crewman is to remain for periods beyond the 45 day period, the effects of the zero-g environment will influence his physiology in unknown ways. The support to that change, whether it be from restraints or specialized foods, cannot at this time be determined. The means to determine such contingencies should be made the subject of extended study for each mission in which this can occur and be related to the choice of the crewman that will be the subject of that specific mission. Although this seems to be basic to the concept, certain relations of man to systems, and particularly to the differences that are probable during rendezvous missions must be carefully studied.

Table 43 shows the schedule of weights and volumes that are anticipated for the missions, which range in length from 45 to 135 days in length.

Table 43. Weights and Volumes

	Man-Days			
	30	45	90	135
Feces 0.33 pound per man-day	9.9 pounds	14.85 pounds	29.7 pounds	44.55 pounds
Urine (based on sample techniques)				
Per man-day 800 (3000*) milliliters Per void 250 (3000*) milliliters	24 liters	36.0 liters	72 liters	108 liters
Sample** volume/ void 75 milliliters***	2.25 liters	3.375 liters	6.750 liters	10.125 liters
Vomit rate once per man per week 4-6 lb ea****	20 pounds	34 pounds	60 pounds	85 pounds

*Abnormal, subject to physiologic definition
**To be collected as individual elements and process-stored
***To be rated at four sample per man-day, subject to post-flight analysis
****Relate to content of stomach

Because of the extensive experiment packages on these 19 missions, it is anticipated that the foods will be stored in coded packages in the external device. Because of such a storage practice, a packaging study is required to define the capability of packaging in the external device under the conditions of the space environment. Secondly, there is the need for consideration of foods transfer to the command module and the appropriate schedule. A detailed coding must be added to the storage concept for purposes of identifying the order in which the foods are to be made available to the crew. In addition, a mission schedule that includes the concept of one (or more) crewman remaining for a 90 or 135 day period, the foods for this man (or men must be supplied. The consideration for such a supply will of necessity be tailored to that man's need. Since this physical condition cannot be totally anticipated, it is difficult to determine, at this time, what the configuration of the food will be or what level of calorie balance he will need.

The influence of the experiment packages that are to be part of the mission must be studied for their influence on the foods management system. If for example, an experiment involves the crew participation in a foods program, the deviations from the normal foods supply must be accounted for. If the foods that are in an experiment package are not totally used or accepted or discarded for any reason, the management of the discard must be defined. Should the foods in an experiment require that the existing foods handling system be in operation, i.e., the use of water, this accommodation must be programmed with the normal supply and/or the emergency supply. Another factor in the foods management will be the relationship of the mission diet as may be related to diets that may be a part of some experimental package. A careful relationship must be made between the experimental diet and the normal spacecraft mission diet so as to provide the balance to the man and still yield the programmed data. In addition, the contingencies for emergency foods, or special diets to meet health degradent problems must be programmed into the experiment and "normal" foods system.

As may be seen from these observations, the influence of the several experiments and the extension of the missions must be rated for both the crew participation and the system accommodation. It is true that a "so-called normal" foods system (i.e., the freeze-dehydrated foods and bite size morsels) can be made available in the proper configuration for the extended missions, but the significant concern will be the deviations from these foods that will be required by the assumed change of man's physiology resulting from the extended duration flights. If the probability of a foods configuration change occurs, i.e., from a semi-solid to a liquid diet, the changes that will be required to accommodate the foods and feeding system

must be made a part of the system design. Table 44 lists the life systems logistics supply for Flights 218 and 219.

Table 44. Life Systems Logistics Supply, Flights 218, 219

Basic Mission 45 days Extended Periods 45, 90, and 135 days (One Man Values Remaining)						
System	Man-Day	45 Man-Days	90* Man-Days Flight 218, 219	135* Man-Days Flight 523	3 Men 45 Days	2 Men Remaining for 90 Days; or 180 Man/Days Flight 523
Food**						
3000 Kcal (lb)	1.5	67.5	135.0	202.5	202.5	270.0
Packaged (lb/cu ft)	2.0 / 0.14ft	90.0 / 12.6ft	180.0 / 25.2ft	270.0 / 37.8ft	270.0 / 37.8ft	360.0 / 50.4ft
* Resupply to be made on the 45-day interval ** Based on Apollo Block II foods concepts caloric density 2000 Kcal/lb						

WATER MANAGEMENT

Management of the water for the missions consists of supplying the required quantity of water in a potable condition to a storage tank as available from the fuel cell operation. The operation of the fuel cell can be rated at a water production rate of approximately 1.5 pounds per hour. Table 45 shows a typical production schedule for a 45-day mission. The use rates shown are minimal with a contingency for periodic use above the normal man-day requirements. As may be seen, the discard values are well above the use requirements. The relationship of a water requirement to an experiment is not yet defined. Since there is ample water available, reasonable quantities of water may be made available to any experiment without detriment to needs of man. However, this requirement must be verified. In addition, storage tanks suitable to experiments must be engineered into the crew systems requirements.

There is no present definition for the water cooled suit. If the water for this suit is derived from available onboard sources, definitions for such water use must be added to this water management schedule.

The acceptable standard for water purity that has been used in the analysis of water systems for space application has been derived from the U.S. Public Health Department. These standards are broad, and are

Table 45. Water Management

Item	Use Requirement (lb)
Water source	
Fuel cell operation	1.5 per hour
Per 24-hour cycle	36
Per 45-day mission	1620
Metabolic Requirements	
Normal duty (three men - 18 pounds)	6 per man-day
Abnormal (experiment duty)* (Two men normal = 12 pounds; one-man experiment nine pounds = day total = 21 pounds	9 per man-day
Mission requirements	
Thirty-eight mission-days 18 pounds	684
Seven mission-days 21 pounds	147
Total	831
Experiment contingency	50
Discard total	739

*Rated for one day per seven-day cycle

applicable to domestic supply and not necessarily the spacecraft. A new definition of space vehicle water standards is required. The sources of contamination on the vehicle arise from an entirely different source, as compared to domestic supply (i.e., the concentrations of gas at reduced pressures, systems cleanliness, etc.). The water from the fuel cell is considered to be mineral- and bacteria-free and, as such, the use to the body may be influenced. If water is to be used in the experiments, the definition of the chemical activity should be studied when this mineral-free

water is used. If the water is to represent earth domestic water, there is the probability that this water should be brought on board at launch, or the capacity be made in the crew systems to store the necessary additives for water stabilization, as may be necessary.

WASTE MANAGEMENT

The Block II Apollo fecal and urine collection systems have not been totally defined. Consequently, it becomes difficult to make assessments of applicability. The concept for fecal collection uses a storage bag collector with a chemical putrefaction control; urine is collected and dumped overboard via a holding tank. A development study is now in progress to define a urine sampling system. The collection of vomit is to be accomplished during shirt sleeve operation by use of a modified fecal collection bag. Methods for collecting vomit while suited are being developed. (Table 42 lists the weights and volumes for a 45-day mission.) It may be noted that all wastes are stored for return to earth during the rendezvous logistics flights.

A program is presently being defined for sampling and study of urine collected during space missions. Valid studies of zero-g effects upon physiological parameters can be derived only from manned orbital flights. Measurements of waste for evidence of bone and muscle decomposition, as well as products of normal metabolism, are generally considered to be accurate indexes of physiological well-being. Both aesthetic and analytical difficulties complicate the acquisition of significant data in these areas under space-flight conditions. The program has been organized to define methods of preserving urine in such a manner as to eliminate biologically and environmentally induced decomposition, obviating the need for on-board analysis and permitting subsequent evaluation upon return to earth.

For the AES missions and experiments, this same program must be further developed to incorporate methods to perform certain analysis onboard the vehicle during the flight. These analyses should include the measure of decalcification and cardiac decomposition, and possible muscular atrophy. The use of microanalytic techniques and recommended and preliminary study of several schemes now operating in laboratory configuration show promise. Several biomedical experiments have anticipated such methods, but did not define the sampling configuration of the fluids (in this case the urine). In addition to medical experiments, there will be the need to collect urine for use in a study of a zero water reclamation system.

Fecal and vomit collection for the AES missions will still be accomplished as indicated for the Block II Apollo. The possible exception will be the accommodation to any experiment program that may use these wastes in some experiment or study.

A new area of concern to the crew and life support systems will be the management of wastes from biological and animal experiments. Until the definition of the type of bioexperiment is engineered and the animal experiment has been further defined, there will be a lack of information for use in systems studies. It is recommended that animal experimentation be accomplished in separate and closed environments relative to the ECS system for manned occupancy. Wastes that will be generated must be isolated and disposed of as soon as possible. This waste not only includes animal excreta, but provision must be made for desposal of sacrificed specimens.

PERSONAL HYGIENE

For the 30- and the 45-day missions, the Apollo Block II systems and system support can be adapted. Table 46 shows the listing of the significant items that comprise the crew systems.

Table 46. Crew System Items

	Man-Days			
	30	45	90	135
Cleaning tissues: 0.15 pound; 0.035 cubic feet	45 pounds	6.75 pounds	13.5 pounds	20.5 pounds
Clothing: one change per 2 days, one pound per change; 0.05 cubic feet	15 pounds	23 pounds	45 pounds	68.0 pounds
Medical and dental care		31.0 pounds 3.3 cubic feet		
Crew system		15.0 pounds 2.5 cubic feet		

The constant wear garment that has been chosen for the Apollo missions has been designed for the crewman who is generally considered a passive passenger in the vehicle. The duty performance of Apollo is light and, therefore, the cleanliness factor of this crewman may be termed light. For the anticipated experimental programs, considerable crew activity is programmed, including extravehicular activity. This activity can be considered medium to heavy, and has been tentatively defined in terms of a heat output value of 12,000 to 18,000 Btu's per man-day. The Apollo heat output for routine flight (light activity) ranged from 11,000 to

12,000 Btu's per day. Apollo estimates for LEM and extravehicular activity have ranged as high as 1600 Btu's per hour. Until significant data is made available, a clothing design to be compatable to the body secretion ratios and the accompanying hygiene pattern cannot be defined. If the present clothing is used, the frequency of change must be increased. If a cooling mechanism in the suit can be made to accommodate the high heat loads, the under garment must be designed to complement the cooling system.

Although no experiment is planned, but has been considered, for a body cleaning system, it is highly desirable to investigate placement of some type body cleaning system on the vehicle. Bag showers and wet suits have been mentioned in the past for the function. These and other concepts must be studied for application, particularly in the 90- and 135-day missions. The body care required for man for missions of this length in the oxygen atmosphere is unknown. Some studies of long-term confinements in pressure suits show concern of the skin irritations. Proper cleaning and body grooming will lessen the possibility of severe medical problems. Some of the irritation may result from the application of the deodorant pad or cream. A revised systems analysis must be programed that will define the required support of body and clothes cleaning concepts, power loading, wastes management, venting, etc.

MEDICAL SUPPORT SYSTEMS

The Block II Apollo first-aid and medical equipment is limited mainly to survival and some preventive medicines. This limitation is based on the light activity that is programmed for the Apollo missions. The support equipment required for the AES missions, particularly for the 90- and 135-day missions, must be carefully prepared. The basic reason for these missions is to evaluate man in the sustained zero-gravity environment. The information that will be required to access man's performance will be the basic part of the several biomedical experiment packages. However, a large portion of the experimentation activity will be performed in a manner postulated to be a general monitor for any space mission. Consequently, the equipment that can support the experimental packages are assumed to be part of the vehicle systems. Careful integration is recommended to avoid unnecessary duplication of systems, operation, and equipments.

Some of the medical experiments that have been postulated show a potential hazard to man's occupancy, or disruption of life equipments. A new survival concept must be adopted that will allow emergency care which may be required as a result of the participation in an experiment. This care concept must be specifically programmed into the monitor for the well-being of the crewman that will remain for the 135-day mission. The

physical well-being of the crewman is being challenged by the mission, consequently, the probability of an interrupted mission program exists. Of significant concern will be the system required to protect the man during Earth reentry.

CONCLUSIONS

Tables 47 and 48 summarize the changes required for each subystem for each configuration. The first table summarizes changes for subsystems in the command and service module; the second table summarizes the subsystems for the external device.

CONFIGURATION 1

Subsystems in the CSM are Block II systems with the exception of communications, environmental control system, and the power system. Communication and data system requires modification to the premodulation processor, the audio center, and the S-band transceiver in order to accommodate wires for transmission and reception of data to and from the external device. In addition, the high-gain antenna would be carried only in sychronous orbit flights, while the rendezvous transponder would only be carried in rendezvous flights. The environmental control system requires the addition of a retractable duct and blower.

The external device requires a power distribution system which would include ac and dc buses. In addition, batteries would be added to complement the power make-up and to handle peak loads. The ECS in the external device requires a thermal control loop, a contaminant control loop, and a repressurization and depressurization system. The power system in the command module must accommodate an ac umbilical from the ac bus. The communication system in the external device requires an audio center and a data-management system to achieve format and condition the experimental data.

CONFIGURATION D'

This configuration contains subsystems in the CSM identical to those of Configuration 1, with the exception of the communication and data system, the earth landing system, and the environmental control system. In addition to the modifications for Configuration 1, the communication and data system requires modification to the PCM telementry unit so that it can accept external device subsystem data. The earth landing system requires the addition of volatile material to the parachute compartment and also modifications to the insulation and external coating of the parachute compartment. The environmental control system must have the same modifications as in Configuration 1. In addition, the compressor, cabin

Table 47. CSM Subsystem Change Summary

Subsystem	Configuration			
	1	D'	D	C
Comm/data	Block II with wiring interface, delete high-gain and rendezvous transponder	Same as 1, plus interface wiring for subsystems data	Same as D'	Same as D
ELS	Block II	Add volatile material	Same as D'	Same as D
ECS	Add blower and duct	Same as 1 plus modified Block II unit with spare compressor and cabin fans plus 100-psi O_2 line	Same as D' plus modified compressor and cabin fans	Same as D less 100-psi O_2
G&N	Block II	Same as 1	Modify IMU, AGC, and ECDU's	Same as D
Power	Block II plus wiring interface	Same as 1	Same as 1	Four 1000-hour cells plus wiring interface
Cryogenic storage	Block II	Same as 1	Same as 1	Four new cryogenic tanks
Propulsion	Block II	Same as 1	Same as 1	Tank size varies per mission requirements
RCS (SM)	Block II	Same as 1	Use LEM tanks (2 sets/quad)	Same as D
SCS	Block II	Same as 1	Modify electronics, add horizon sensors, add redundant units	Same as D

Table 48. External Device Subsystem Summary

Subsystem	Configuration			
	1	D'	D	C
Comm/data	Add audio CTR and data management system	Same as 1, plus signal conditioner for subsystems data	Same as D'	Same as 1
ELS	—	—	—	—
ECS	Thermal control loop and repression system	Same as 1	Thermal control loop	Same as D
G&N	—	—	—	—
Power	Battery for make-up and peaks, plus power distribution system	Power distribution system plus battery for peaks plus three 400-hour fuel cells plus cooling loop	Power distribution system plus battery for peaks plus three 1000-hour cells plus cooling loop	Power distribution system plus battery for peaks
Cryogenic storage	None	7 Block II tanks	4 new cryogenic tanks	None
Propulsion	—	—	—	—
RCS	—	—	—	—
SCS	—	—	—	—

fan, and other items must be made redundant. Also, a 100-psi O_2 line which conducts oxygen from the external device is interfaced with the command module system.

The subsystems on the external device are the same as in Configuration 1, with the exception of the cryogenics system. The communication system on the external device requires a signal-conditioning unit to condition the subsystem data from the external device subsystems. A power distribution system, similar to that of Configuration 1 is required, and also sufficient batteries to handle peak loads. Needed are three 400-hour fuel cells and a cooling loop with radiators. Seven Block II tanks must be installed in the external device to supply reactant to the power system and provide oxygen for the crew.

CONFIGURATION D

The changes to the command module subsystems are the same as those for Configuration D' or 1, with the exception of the environmental control system, the guidance and navigation system, the service module RCS, and the SCS. In the environmental control system, the compressor bearings and cabin-fan bearings are modified to sustain longer mission life; also, some small components are made redundant. The guidance and navigation system requires modification to the inertial measurement unit, the guidance computer, and the coupling display unit. This must be done to allow some portions of the system to be turned off when not in use. Also, modifications are needed to accomplish the mission requirements, such as stringent attitude hold or local vertical hold for long periods of time. The reaction control system in the service module uses LEM tankage (two sets per quad) to accomplish the experimental requirements. The stabilization and control system requires redundancies and the addition of horizon sensors and modification to some electronic units so it can accept the horizon sensor outputs. The addition of the horizon sensors provide a local, vertical-hold capability.

The subsystems in the external device are the same as Configuration D' with the exception of ECS, power, and cryogenic storage systems. The environmental control system requires only a thermal control loop and no repressurization system because this is provided by the cryogenic storage system. The power system requires the power distribution system and batteries as in Configuration D'. However, three 1000-hour fuel cells are carried plus a cooling loop and radiator. Since the Apollo X type tanks are carried for the cryogenic storage system, fewer tanks are required.

CONFIGURATION C

The subsystems in the command and service module are the same as those for Configuration D, with the exception of the environmental control system, power system, cryogenic system, and propulsion system. The environmental control system is the same as in Configuration D with the exception of the 100-psi oxygen line. This line is deleted because no cryogenic fuel is carried in the external device. Otherwise, it incorporates the same modification as were required for Configuration D. The power system uses four 1000-hour fuel cells and must accommodate the same wiring interface as in Configuration 1. To supply reactant for the fuel cells and oxygen for the crew, Apollo X type tanks are carried in Sectors 1 and 4 of the service module. The propulsion system uses smaller tank sizes than Block II. It appears that tanks of 10,000 total capacity will suffice for all missions except those for which Block II tanks are required. The subsystems in the external device are the same as in Configuration D except for the deletion of the fuel cells and cooling loop in the power system, deletion of cryogenic tankage, and the deletion of the signal-conditioning unit for subsystem data.

APPENDIX A

RCS-SCS MISSION DATA

This appendix documents the results of the analyses conducted during the study program. The material has been organized by flight number. Each flight or mission package contains tabulations of control requirements, rotational maneuvers, operating modes, RCS requirements, and a system summary. Specific problems were encountered in Flights 215 and 221 because of the artifical gravity rotational experiment, on Flight 518 because of the solar panels, and on Flight 507 because of guidance system reliability considerations. These problems are discussed in the body of the report.

FLIGHT 209

Mission duration: 14 days. Orbit inclination: 28.5 degrees. Orbit altitude: 200 nautical miles. Crew size: 3 men.

Table 49. Control Requirements Summary, SCS, Flight 209

Experiment Number	Axis	Orientation Accuracy (Degrees)	Angular Velocity Limit (deg/sec)	Angular Acceleration Limit (deg/sec/sec)	Time Per Mission (Hours)	Operating Mode
0601	All	NA	±0.2*	NA	3.5	SCS rate control
1503	All	±5	±0.2	NA	1.9	SCS attitude hold with manual maneuver
1502	All	±5	±0.2	NA	1.9	SCS attitude hold with manual maneuver
1501	All	±5	±0.2	NA	1.9	SCS attitude hold with manual maneuver
0101-0102 Zero-G mode	All	NA	Minimum available	NA	1.0	SCS rate control (minimum impulse)
0101-0102 Rotation mode	All	NA	See profile	NA	2.5	SCS rotation
0101-0102 Linear acceleration mode	All	NA	Minimum	NA	0.5	SCS TVC

(*) Actual requirement stated as 10^{-4} g equivalent
NA - Not applicable

Table 50. Rotational Maneuver Summary, Flight 209

Experiment Number	Axis	Slewing Rate (Degrees Per Second)	Maneuver Mode		Number Per Mission
			Manual	Automatic	
0101-0102	Pitch or yaw	Accelerate to 9 RPM in steps of 3 RPM each, and decelerate in similar manner	✓		6
1501*	Pitch and yaw	0.5	✓		8
1502*	Pitch and yaw	0.5	✓		8
1503*	Pitch and yaw	0.5	✓		8
(*)Approximately four maneuvers per hour of operation are assumed necessary to allow observation of EVA from vehicle interior					

Table 51. Operating Mode, Time Summary, Flight 209

Mode of Operation	Time for Ascent (Hours)	Time on Orbit (Hours)	Time for Entry (Hours)	Total Time (Hours)
Free drift		319.3		319.3
SCS attitude hold (minimum impulse)		4.5		4.5
SCS TVC		0.5		0.5
SCS rotation		2.5		2.5
Ascent	0.2			0.2
Entry			0.3	0.3
SCS attitude hold and manual maneuver	1.5	5.7	1.5	8.7
Total time				336.0

Table 52. SCS System Summary, Flight 209

Component	Weight Per Unit Pounds	Volume Per Unit (Cu In)	Number for Basic System	Number of Spares	Weight Per System (Pounds)	Volume Per System (Cu In)
Solenoid driver assembly	12.2	476	1	0	12.2	476
Control electronics assembly	16.1	495	1	0	16.1	495
Display electronics assembly	19.6	615	1	0	19.6	615
Gyro display coupler	15.7	430	1	0	15.7	430
Servo amplifier (TVC)	14.6	430	1	0	14.6	430
Gyro assembly	17.5	525	2	0	35.0	1050
Flight director attitude indicator	7.9	428	2	0	15.8	856
Gimbal position and fuel pressure indicator	3.7	150	1	0	3.7	150
Attitude set control panel	3.7	115	1	0	3.7	115
Translation control	4.3	70	1	0	4.3	70
Rotation control	6.2	94	2	0	12.4	188
Entry monitor system	5.0	112	1	0	5.0	112
Totals					158.1	4987

Modifications: none

NOTES: Reliability of basic system = 0.9976

Table 53. Service Module—RCS Requirements, Flight 209

Mission \ Requirement	Maximum Burn Time per Jet (Second)			Maximum Number Starts per Jet			Propellant Weight (Pounds)		
	Roll	Pitch	Yaw	Roll	Pitch	Yaw	Roll	Pitch	Yaw
Ascent	4.5	37.8	49.5	140	115	115	4.3	42.2	43.4
Orbit-keeping	0	7.4	7.4				0	11.0	11.0
In-orbit	0.7	18.5	18.5	517	593	593	1.0	273.0	273.0
Pre-entry	1.6	12.5	12.1	180	133	133	2.0	9.3	8.5
Totals	6.8	76.2	87.5	837	841	841	7.3	335.5	335.9

FLIGHT 211

Mission duration: 30 days. Orbit inclination: 28.5 degrees.
Orbit altitude: 200 nautical miles. Crew size: 3 men.

Table 54. Control Requirements Summary, SCS, Flight 211.

Experiment Number	Axis	Orientation Accuracy (Degrees)	Angular Velocity Limit (deg/sec)	Angular Acceleration Limit (deg/sec/sec)	Time Per Mission (Hours)	Operating Mode
0904 *	All	±0.5	±0.04	NA	23.3	G&N primary
0904′	All	±5.0	±0.2	NA	0.17	SCS attitude hold
0101-0102 Zero-G mode	All	NA	Minimum available	NA	1.0	SCS rate control (minimum impulse)
0101-0102 Rotation mode	All	NA	See profile	NA	2.5	SCS rotation
0101-0102 Linear acceleration mode	All	NA	Minimum	NA	0.5	SCS TVC

NA - Not applicable
* - X axis along local horizontal and in plane of orbit * * *

Table 55. Rotational Maneuver Summary, Flight 211

Experiment Number	Axis	Slewing Rate (Degrees Per Second)	Maneuver Mode		Number Per Mission
			Manual	Automatic	
0904	3 axis	0.5		√	28
	roll	0.5		√	84
0101-0102	Pitch or yaw	Accelerate to 9 RPM in steps of 3 RPM each and decelerate in similar manner	√		12

Table 56. Operating Mode Summary, Flight 211

Mode of Operation	Time for Ascent (Hours)	Time on Orbit (Hours)	Time for Entry (Hours)	Total Time (Hours)
Free Drift				685.0
G&N Primary		23.3		23.3
SCS attitude hold (minimum impulse)		2.2		2.2
SCS attitude hold and manual maneuver	1.5		1.5	3.0
SCS Rotation		5.0		5.0
SCS TVC		1.0		1.0
Ascent	0.2			0.2
Entry			0.3	0.3
Total				720.0

Table 57. SCS System Summary, Flight 211

Component	Weight Per Unit (Pounds)	Volume Per Unit (Cubic Inches)	Number for Basic System	Number of Spares	Weight Per System (Pounds)	Volume Per System (Cu. In.)
Solenoid driver assembly	12.2	476	1	0	12.2	476
Control electronics assembly	16.1	495	1	0	16.1	495
Display electronic assembly	19.6	615	1	0	19.6	615
Gyro display coupler	15.7	430	1	0	15.7	430
Servo amplifier (TVC)	14.6	430	1	0	14.6	430
Gyro assembly	17.5	525	2	0	35.0	1050
Flight director attitude indicator	7.9	428	2	0	15.8	856
Gimbal position and fuel pressure indicator	3.7	150	1	0	3.7	150
Attitude set control panel	3.7	115	1	0	3.7	115
Translation control	4.3	70	1	0	4.3	70
Rotation control	6.2	94	2	0	12.4	188
Entry monitor system	5.0	112	1	0	5.0	112
Totals					158.1	4987

Modifications: None

NOTE: Basic system reliability = 0.9984

Table 58. Service Module—RCS Requirements, Flight 211

Requirement / Mission Phase	Maximum Burn Time per Jet (Second)			Maximum Number Starts per Jet			Propellant Weight (Pounds)		
	Roll	Pitch	Yaw	Roll	Pitch	Yaw	Roll	Pitch	Yaw
Ascent	4.6	39.0	51.0	140	115	115	6.3	42.5	43.7
Orbit-keeping	0	16.0	16.0				0	23.5	23.5
In-orbit	28.5	210.0	151.0	2712	923	923	42.2	311.0	223.0
Pre-entry	1.6	12.7	12.3	180	133	133	2.2	9.4	8.6
Totals	34.7	277.7	230.3	2906	1171	1171	50.7	386.4	298.8

FLIGHT 507

Mission duration: 14 days. Orbit inclination: Polar. Orbit altitude: 200 nautical miles. Crew size: 3 men.

Table 59. Control Requirements Summary, SCS, Flight 507

Experiment Number	Axis	Orientation Accuracy (Degrees)	Angular Velocity Limit (deg/sec)	Angular Acceleration Limit (deg/sec/sec)	Time Per Mission (Hours)	Operating Mode
1201	All*	±35	±0.2*	No requirement	96	G&N prime
1202	All	±2	±0.2*	No requirement	0.33	G&N prime
1405	All	±5	DNA	NA	3	SCS attitude hold
0901 ** 0902	All	±0.1	±0.01	No requirement	31.5	G&N prime manual with maneuver
0101-0102 Zero-G mode	All	NA	Minimum available	NA	1.0	SCS rate control (minimum impulse)
0101-0102 Rotation mode	All	NA	See profile	NA	2.5	SCS rotation
0101-0102 Linear acceleration mode	All	NA	Minimum	NA	0.5	SCS TVC

(*) NAA assumed (Not specified)
(**) All 1901 and 1902 experiments assumed to be run together.
NA - Not applicable

Table 60. Rotational Maneuver Summary, Flight 507

Experiment Number	Axis	Slewing Rate (Degrees Per Second)	Maneuver Mode		Number Per Mission
			Manual	Automatic	
1405	R & Y or P	0.5*		✓	14
1202	3	0.5*		✓	4
1201	3	0.5*		✓	4
0101-0102	Pitch or yaw	Accelerate to 9 RPM in steps of 3 RPM each, and decelerate in similar manner	✓		6
0901** and 0902	3	0.5*		✓	20
0901 and 0902	3	0.5*	✓		40

(*)NAA assumed (no requirement specified)
(**)Based on one maneuver to local vertical each 11 orbits

Table 61. Operating Mode - Time Summary, Flight 507

Mode of Operation	Time for Ascent (Hours)	Time on Orbit (Hours)	Time for Entry (Hours)	Total Time (Hours)
Ascent	0.2			0.2
SCS attitude hold and manual maneuvers	1.5		1.5	3.0
SCS TVC		0.5		0.5
SCS attitude hold (minimum impulse)		4.0		4.0
SCS rotation		2.5		2.5
G&N prime		96.3		96.3
G&N prime with manual maneuvers		31.5		3.15
Entry			0.3	0.3
Free drift		197.7		197.7
Total time				336.0

Table 62. SCS System Summary, Flight 507

Component	Weight Per Unit (Pounds)	Volume Per Unit (Cubic Inches)	Number for Basic System	Number of Spares	Weight Per System (Pounds)	Volume Per System (Cu. In.)
Solenoid driver assembly	12.2	476	1	-	12.2	476
Control electronics assembly	16.1	495	1	-	16.1	495
Display electronics assembly	19.6	615	1	-	19.6	615
Gyro display coupler	15.7	430	1	-	15.7	430
Servo amplifier (TVC)	14.6	430	1	-	14.6	430
Gyro assembly	17.5	525	2	-	35.0	1050
Flight director attitude indicator	7.9	428	2	-	15.8	856
Gimbal position and fuel pressure indicator	3.7	150	1	-	3.7	150
Attitude set control panel	3.7	115	1	-	3.7	115
Translation control	4.3	70	1	-	4.3	70
Rotation control	6.2	94	2	-	6.2	94
Entry monitor system	5.0	112	1	-	5.0	112
Totals					158.1	4987

Modifications: None

NOTE: Reliability of basic system = 0.9897

Table 63. Service Module—RCS Summary, Flight 507

Requirement / Mission Phase	Maximum Burn Time per Jet (Second)			Maximum Number Starts per Jet			Propellant Weight (Pounds)		
	Roll	Pitch	Yaw	Roll	Pitch	Yaw	Roll	Pitch	Yaw
Ascent	5.3	44.5	58.5	140	115	115	9.9	47.6	48.8
Orbit-keeping	0	7.4	7.4				0	11.0	11.0
In-orbit	84.5	162.0	162.0	5882	2660	2660	125.2	240.5	240.5
Pre-entry	1.6	12.7	12.3	180	133	133	2.2	9.4	8.6
Totals	91.4	226.6	240.2	6202	2908	2908	137.3	308.5	308.9

FLIGHT 509

Mission duration: 14 days. Orbit inclination: 0 degrees.
Orbit altitude: 300 nautical miles. Crew size: 2 men.

Table 64. Control Requirements Summary, SCS, Flight 509

Experiment Number	Axis	Orientation Accuracy (Degrees)	Angular Velocity Limit (deg/sec)	Angular Acceleration Limit (deg/sec/sec)	Time Per Mission (Hours)	Operating Mode
0101-0102 Zero-G mode	All	NA	Minimum	NA	0.67	SCS rate control (minimum impulse)
0101-0102 Rotation mode	All	NA	See profile	NA	1.67	SCS rotation
0101-0102 Linear acceleration mode	All	NA	Minimum	NA	0.33	SCS TVC
0801	All	±0.5*	±0.2*	No requirement	7	G&N prime
1603-1601	All	±0.5*	±0.2*	No requirement	3.5	G&N prime with manual maneuvers
1306 Erect	All	±5	±0.2*	No requirement	0.5	SCS attitude hold with manual maneuvers
1306 Test	All	±0.3	±0.02*	No requirement	6	G&N prime
1304 Deploy	All	±0.5*	±0.2*	No requirement	1	SCS attitude hold with manual maneuvers
1304 Test	All	±0.5*	±0.2*	No requirement	2	SCS attitude hold with manual maneuvers
0705B	All	±2	±0.2*	No requirement	0.9	G&N prime
0705B, 1303	All	±0.5*	±0.2*	No requirement	0.2	G&N prime
1302, 1303	All	±5	±0.2	No requirement	4	SCS attitude hold with manual maneuvers

(*)NAA assumed, no requirement specified
NA - Not applicable

Table 65. Rotational Maneuver Summary, Flight 509

Experiment Number	Axis	Slewing Rate (Degrees Per Second)	Maneuver Mode		Number Per Mission
			Manual	Automatic	
0101-0102	Pitch or yaw	Accelerate to 9 RPM in steps of 3 RPM each, and decelerate in similar manner	✓		4
0801	Roll and pitch or yaw	0.5*		✓	14
1603-1601	3 axis	0.5*	✓		6
1306 Erect	3 axis	0.5*	✓		1
1306 Test	Roll and pitch or yaw	0.5*		✓	12
1304 Deploy	3 axis	0.5*	✓		4
1304 Test	Roll and pitch or yaw	0.5*	✓		8
1302-1303	3 axis	0.5*	✓		6
0705B, 1303	3 axis	0.5*		✓	1
0705B	Roll and pitch or yaw	0.5*		✓	5
*NAA assumed					

Table 66. Operating Mode, Time Summary, Flight 509

Mode of Operation	Time for Ascent (Hours)	Time on Orbit (Hours)	Time for Entry (Hours)	Total Time (Hours)
Ascent	0.2			0.2
SCS attitude hold with manual maneuvers	1.5	7.5	1.5	10.5
SCS TVC		0.3		0.3
SCS rotation		1.7		1.7
SCS attitude hold (minimum impulse)		0.7		0.7
G&N prime		14.1		14.1
G&N prime with manual maneuvers		3.5		3.5
Entry			0.3	0.3
Free drift		304.7		304.7
Total time				336.0

Table 67. SCS System Summary, Flight 509

Component	Weight Per Unit (Pounds)	Volume Per Unit (Cubic Inches)	Number for Basic System	Number of Spares	Weight Per System (Pounds)	Volume Per System (Cu. In.)
Solenoid driver assembly	12.2	476	1	-	12.2	476
Control electronics assembly	16.1	495	1	-	16.1	495
Display electronics assembly	19.6	615	1	-	19.6	615
Gyro display coupler	15.7	430	1	-	15.7	430
Servo amplifier (TVC)	14.6	430	1	-	14.6	430
Gyro assembly	17.5	525	2	-	35.0	1050
Flight director attitude indicator	7.9	428	2	-	15.8	856
Gimbal position and fuel pressure indicator	3.7	150	1	-	3.7	150
Attitude set control panel	3.7	115	1	-	3.7	115
Translation control	4.3	70	1	-	4.3	70
Rotation control	6.2	94	2	-	6.2	94
Entry monitor system	5.0	112	1	-	5.0	112
Totals					158.1	4987

Modifications: None

NOTES: Reliability of basic system = 0.9973

Table 68. Service Module—RCS Requirements, Flight 509

Mission Phase / Requirement	Maximum Burn Time per Jet (Second)			Maximum Number Starts per Jet			Propellant Weight (Pounds)		
	Roll	Pitch	Yaw	Roll	Pitch	Yaw	Roll	Pitch	Yaw
Ascent	4.7	39.5	51.8	140	115	115	10.3	41.0	42.6
Orbit-keeping	0	0	0	0	0	0	0	0	0
In-orbit	27.1	242.0	154.0	1501	1178	1178	40.5	359.0	228.0
Pre-entry	1.7	13.5	13.0	180	133	133	2.8	9.8	8.8
Totals	33.5	295.0	218.8	1821	1426	1426	53.6	409.8	279.4

FLIGHT 215

Mission duration: 14 days. Orbit inclination: 50 degrees. Orbit altitude: 200 nautical miles. Crew size: 3 men.

Table 69. Control Requirements Summary, SCS, Flight 215

Experiment Number	Axis	Orientation Accuracy (Degrees)	Angular Velocity Limit (deg/sec)	Angular Acceleration Limit (deg/sec/sec)	Time Per Mission (Hours)	Operating Mode
0901 and 0902	All	±0.1	±0.01	No requirement	78	G&N prime with manual maneuver
0301 rotation	All	NA	See profile	NA	16.5	SCS rotation
0301 Zero-G	All	NA	Minimum	NA	1.5	SCS attitude hold (minimum impulse)
0101-0102 Zero-G mode	All	NA	Minimum available	NA	1.0	SCS rate control (minimum impulse)
0101-0102 rotation mode	All	NA	See profile	NA	2.5	SCS rotation
0101-0102 Linear acceleration mode	All	NA	Minimum	NA	0.5	SCS TVC
NA - Not applicable						

Table 70. Rotational Maneuver Summary, Flight 215

Experiment Number	Axis	Slewing Rate (Degrees Per Second)	Maneuver Mode		Number Per Mission
			Manual	Automatic	
0901 and 0902	3	0.5		✓	52
0901 and 0902	3	0.5	✓		10
0101-0102	Pitch or yaw	Accelerate to 9 RPM in steps of 3 RPM each, and decelerate in similar manner	✓		6
0301	Yaw		✓		1 Sequence

Table 71. Operating Mode - Time Summary, Flight 215

Mode of Operation	Time for Ascent (Hours)	Time on Orbit (Hours)	Time for Entry (Hours)	Total Time (Hours)
Ascent	0.2			0.2
SCS attitude hold (minimum impulse)		2.5		2.5
SCS rotation		19.0		19.0
SCS TVC		0.5		0.5
G&N, prime with manual maneuver		78.0		78.0
SCS attitude hold with manual maneuver	1.5		1.5	3.0
Free drift		232.5		232.5
Entry			0.3	0.3
Total time				336.0

Table 72. SCS Systems Summary, Flight 215

Component	Weight Per Unit (Pounds)	Volume Per Unit (Cu In)	Number for Basic System	Number of Spares	Weight Per System (Pounds)	Volume Per System (Cu In)
Solenoid driver assembly	12.2	476	1	-	12.2	476
Control electronics assembly	16.1	495	1	-	16.1	495
Display electronics assembly	19.6	615	1	-	19.6	615
Gyro display coupler	15.7	430	1	-	15.7	430
Servo amplifier (TVC)	14.6	430	1	-	14.6	430
Gyro assembly	17.5	525	2	-	35.0	1050
Flight director attitude indicator	7.9	428	2	-	15.8	856
Gimbal position and fuel pressure indicator	3.7	150	1	-	3.7	150
Attitude set control panel	3.7	115	1	-	3.7	115
Translation control	4.3	70	1	-	4.3	70
Rotation control	6.2	94	2	-	6.2	94
Entry monitor system	5.0	112	1	-	5.0	112
Totals					158.1	4987

Modifications: None

NOTE: Reliability of basic system = 0.9966.

Table 73. Service Module—RCS Requirements, Flight 215

Mission Phase \ Requirement	Maximum Burn Time per Jet (Second)			Maximum Number Starts per Jet			Propellant Weight (Pounds)		
	Roll	Pitch	Yaw	Roll	Pitch	Yaw	Roll	Pitch	Yaw
Ascent	7.8	65.0	86.0	140	115	115	36.0	65.0	65.0
Orbit-keeping	0	7.4	7.4					11.0	11.0
In-orbit	303	44.5	44.5	8446	2439	2439	178.5	200.7	200.7
Pre-entry	1.7	13.1	12.7	180	133	133	2.5	9.6	8.7
Totals	312.5	130.0	150.6	8766	2687	2687	217.0	286.3	285.4

FLIGHT 513

Mission duration: 14 days. Orbit inclination: 81.5 degrees. Orbit altitude: 200 to 700 nautical miles. Crew size: 3 men.

Table 74. Control Requirements Summary, SCS, Flight 513

Experiment Number	Axis	Orientation Accuracy (Degrees)	Angular Velocity Limit (deg/sec)	Angular Acceleration Limit (deg/sec/sec)	Time Per Mission (Hours)	Operating Mode
0802A	All	±0.5	±0.5	0.5	18.7	G&N prime with manual maneuvers
0802B	All	±0.5	±0.5	0.5	28.0	G&N prime with manual maneuvers
0802C	All	±0.15	±0.05*	No requirement	7	G&N prime with manual maneuvers
0802D	All	±0.5	±1.0	No requirement	0.5	G&N prime
0502	All	±0.5	±0.2*	No requirement	7	G&N prime
0503 (Release)	All	±2.0	±0.2*	No requirement	Negligible	SCS attitude hold
0503 (Observation)	All	±5.0	±0.2*	No requirement	5	SCS attitude hold
1507	All	±5.0	±0.5	0.5	10	SCS attitude hold with manual maneuvers
1602 (Transfer)	All	Normal system operation			2.0	G&N prime attitude hold and TVC
1602 (Observation)	All	±5	±0.04	0.24	0.1	G&N prime with manual maneuvers
0101-0102 Zero-G mode	All	NA	Minimum available	NA	1.0	SCS rate control (minimum impulse)
0101-0102 Rotation mode	All	NA	See profile	NA	2.5	SCS rotation
0101-0102 Linear acceleration mode	All	NA	Minimum	NA	0.5	SCS TVC

(*) NAA assumed, no requirement specified
NA - Not applicable

Table 75. Rotational Maneuver Summary, Flight 513

Experiment Number	Axis	Slewing Rate (Degrees Per Second)	Maneuver Mode		Number Per Mission
			Manual	Automatic	
0101-0102	Pitch or yaw	Accelerate to 9 RPM in steps of 3 RPM each, and decelerate in similar manner	✓		6
0802A	3 axis	0.5*	✓	✓	112
0802B	3 axis	0.5*	✓	✓	168
0802C	3 axis	0.5*	✓	✓	56
0802D	3 axis	0.5*	✓		7
0503**	3 axis	0.5*	✓		3
0502	3 axis	0.5*		✓	28
1507	Pitch and yaw	0.5*	✓		40
1602	3 axis	0.5*	✓	✓	4

(*)NAA assumed
(**)One reorientation to establish suitable location for each of 3 cloud releases

Table 76. Operating Mode, Time Summary, Flight 513

Mode of Operation	Time for Ascent (Hours)	Time on Orbit (Hours)	Time for Entry (Hours)	Total Time (Hours)
Ascent	0.2			0.2
SCS attitude hold with manual maneuvers	1.5	10.0	1.5	13.0
SCS attitude hold (minimum impulse)		6.0		6.0
G&N prime		7.5		7.5
G&N prime with manual maneuvers		53.8		53.8
G&N prime (attitude hold and TVC)		2.0		2.0
SCS TVC		0.5		0.5
SCS rotation		2.5		2.5
Entry			**0.3**	**0.3**
Free drift		**250.2**		**250.2**
Total time				**336.0**

Table 77. SCS System Summary, Flight 513

Component	Weight Per Unit (Pounds)	Volume Per Unit (Cubic Inches)	Number for Basic System	Number of Spares	Weight Per System (Pounds)	Volume Per System (Cu. In.)
Solenoid driver assembly	12.2	476	1	-	12.2	476
Control electronics assembly	16.1	495	1	-	16.1	495
Display electronics assembly	19.6	615	1	-	19.6	615
Gyro display coupler	15.7	430	1	-	15.7	430
Servo amplifier (TVC)	14.6	430	1	-	14.6	430
Gyro assembly	17.5	525	2	-	35.0	1050
Flight director attitude indicator	7.9	428	2	-	15.8	856
Gimbal position and fuel pressure indicator	3.7	150	1	-	3.7	150
Attitude set control panel	3.7	115	1	-	3.7	115
Translation control	4.3	70	1	-	4.3	70
Rotation control	6.2	94	2	-	6.2	94
Entry monitor system	5.0	112	1	0	5.0	112
Totals					158.1	4987

Modifications: None

NOTE: Reliability of basic system = 0.9949

Table 78. Service Module—RCS Requirements, Flight 513

Mission Phase / Requirement	Maximum Burn Time per Jet (Second)			Maximum Number Starts per Jet			Propellant Weight (Pounds)		
	Roll	Pitch	Yaw	Roll	Pitch	Yaw	Roll	Pitch	Yaw
Ascent	4.9	41.0	53.5	140	115	115	9.3	43.4	44.5
Orbit-keeping	0	7.4	7.4				0	11.0	11.0
In-orbit	185.0	128.0	128.0	9942	7149	7149	274.0	189.5	189.5
Pre-entry	1.6	12.6	12.2	180	133	133	2.1	9.3	8.6
Totals	191.5	189.0	201.1	10262	7397	7397	285.4	253.2	253.6

FLIGHT 219

Mission duration: 45 days. Orbit inclination: 28.5 degrees. Orbit altitude: 200 nautical miles. Crew size: 3 men.

Table 79. Control Requirements Summary, SCS, Flight 219

Experiment Number	Axis	Orientation Accuracy (Degrees)	Angular Velocity Limit (deg/sec)	Angular Acceleration Limit (deg/sec/sec)	Time Per Mission (Hours)	Operating Mode
0101-0102 Zero-G	All	NA	Minimum	NA	3.2	SCS rate control (minimum impulse)
0101-0102 Rotation	All	NA	See profile	NA	8.1	SCS rotation
0101-0102 Linear acceleration	All	NA	Minimum	NA	1.6	SCS TVC
1501-1502	All	±5	±0.2	NA	5.2	SCS attitude hold with manual maneuver
1503-1504	All	±5	±0.2	NA	5.2	SCS attitude hold with manual maneuver
1505	All	±5	±0.2	NA	5.2	SCS attitude hold with manual maneuver
1504	All	±5	±0.2	NA	9.5	SCS attitude hold with manual maneuver
1407	Pitch yaw roll	±0.5 ±0.5 ±10	±0.02 ±0.02 ±0.02	NA NA NA	9	G&N prime and manual and scanners
0601	All	NA	±0.2 *	NA	11.2	SCS rate control (minimum impulse)
0602A	All	NA	±0.2 *	10^{-4}	22.5	SCS rate control (minimum impulse)
0602B	All	NA	±0.2 **	10^{-4}	6	SCS rate control (minimum impulse)
0603A	All	NA	±0.2 *	10^{-4}	2.5	SCS rate control (minimum impulse)

(*) Actual requirement stated at 10^{-4} g equivalent
(**) NAA assumed, no requirement specified
(***) Solar orientation
NA - Not applicable

Table 79. Control Requirements Summary, SCS, Flight 219 (Cont)

Experiment Number	Axis	Orientation Accuracy (Degrees)	Angular Velocity Limit (deg/sec)	Angular Acceleration Limit (deg/sec/sec)	Time Per Mission (Hours)	Operating Mode
0604	All	NA	±0.2 **	Minimal **	4	SCS rate control (minimum impulse)
0605	All	±0.5 ***	±0.2	Minimal **	11.2	SCS attitude hold (minimum impulse)
0606	Assume will not schedule					

(*) Actual requirement stated at 10^{-4} g equivalent
(**) NAA assumed, no requirement specified
(***) Solar orientation
NA - Not applicable

Table 80. Rotational Maneuver Summary, Flight 219

Experiment Number	Axis	Slewing Rate (Degrees Per Second)	Maneuver Mode		Number Per Mission
			Manual	Automatic	
0101-0102	Pitch or yaw	Accelerate to 9 RPM in steps of 3 RPM each and decelerate in similar manner	✓		19
1501-1502	Pitch and yaw	0.5[1]	✓		20
1503-1504	Pitch and yaw	0.5[1]	✓		20
1505	Pitch and yaw	0.5[1]	✓		20
1504	Pitch and yaw	0.5[1]	✓		38
1407 Landmark	3 axis	0.5[1]	✓		24[2]
1407 Stellar	3 axis	0.5[1]	✓		18[3]
1407 Horizon and Doppler Alt.	3 axis	0.5[1]	✓		6[4]
0605	3 axis	0.5[1]	✓		22

[1] NAA assumed
[2] Four sightings per orbit for 6 orbits
[3] Three sightings per orbit for 6 orbits
[4] Sightings are grouped such that scanners must only be oriented once per orbit for 6 orbits

Table 81. Operating Mode, Time Summary, Flight 219

Mode of Operation	Time for Ascent (Hours)	Time on Orbit (Hours)	Time for Entry (Hours)	Total Time (Hours)
Ascent	0.2			0.2
SCS attitude hold (minimum impulse)		60.6		60.6
SCS attitude hold with manual maneuvers	1.5	25.1	1.5	28.1
SCS rotation		8.1		8.1
SCS TVC		1.6		1.6
G&N prime with manual maneuvers and scanners		9.0		9.0
Free drift		972.1		972.1
Entry			0.3	0.3
Total time				1080.0

Table 82. SCS System Summary, Flight 219

Component	Weight Per Unit (Pounds)	Volume Per Unit (Cubic Inches)	Number for Basic System	Number of Spares	Weight Per System (Pounds)	Volume Per System (Cu. In.)
Solenoid driver assembly	12.2	476	1	-	12.2	476
Control electronics assembly	16.1	495	1	-	16.1	495
Display electronics assembly	19.6	615	1	-	19.6	615
Gyro display coupler	15.7	430	1	-	15.7	430
Servo amplifier (TVC)	14.6	430	1	-	14.6	430
Gyro assembly	17.5	525	2	-	35.0	1050
Flight director attitude indicator	7.9	428	2	-	15.8	856
Gimbal position and fuel pressure indicator	3.7	150	1	-	3.7	150
Attitude set control panel	3.7	115	1	-	3.7	115
Translation control	4.3	70	1	-	4.3	70
Rotation control	6.2	94	2	-	6.2	94
Entry monitor system	5.0	112	1	0	5.0	112
Totals					158.1	4987

Modifications: None

NOTE: Reliability of basic system = 0.9817

Table 83. Service Module - RCS Requirements, Flight 219C

Mission Phase \ Requirement	Maximum Burn Time per Jet (Second)			Maximum Number Starts per Jet			Propellant Weight (Pounds)		
	Roll	Pitch	Yaw	Roll	Pitch	Yaw	Roll	Pitch	Yaw
Ascent	4.8	40.0	52.5	140	115	115	7.0	43.5	44.9
Orbit-keeping	0	24.0	24.0				0	35.5	35.5
In-orbit	72.0	645.0	580.0	3814	3183	3180	107.1	957.5	857.5
Pre-entry	1.7	13.0	12.5	180	133	133	2.3	9.6	8.7
Totals	78.5	722.0	669.0	4135	3431	3428	116.4	1046.1	1024.9

Table 84. Service Module - RCS Requirements, Flight 219D

Mission Phase \ Requirement	Maximum Burn Time per Jet (Second)			Maximum Number Starts per Jet			Propellant Weight (Pounds)		
	Roll	Pitch	Yaw	Roll	Pitch	Yaw	Roll	Pitch	Yaw
Ascent	4.7	39.5	51.8	140	115	115	5.9	43.0	44.7
Orbit-keeping	0	24.0	24.0				0	35.5	35.5
In-orbit	87.0	655.0	655.0	3746	3487	3484	128.9	973.5	973.5
Pre-entry	1.8	13.7	13.3	180	133	133	3.0	9.9	8.9
Totals	93.5	732.2	744.1	4066	3735	3732	137.8	1061.9	1062.6

FLIGHT 221

Mission duration: 45 days. Orbit inclination: 28.5 degrees. Orbit altitude: 200 nautical miles. Crew size: 3 men.

Table 85. Control Requirements Summary, SCS, Flight 221

Experiment Number	Axis	Orientation Accuracy (Degrees)	Angular Velocity Limit (deg/sec)	Angular Acceleration Limit (deg/sec/sec)	Time Per Mission (Hours)	Operating Mode
0301	All	NA	See profile	NA	1069	SCS rotation
NA - Not applicable						

Table 86. Rotational Maneuver Summary, Flight 221

Experiment Number	Axis	Slewing Rate (Degrees Per Second)	Maneuver Mode		Number Per Mission
			Manual	Automatic	
0301	Pitch or yaw	See profile	✓		One sequence

Table 87. Operating Mode, Time Summary, Flight 221

Mode of Operation	Time for Ascent (Hours)	Time on Orbit (Hours)	Time for Entry (Hours)	Total Time (Hours)
Ascent	0.2			0.2
SCS attitude hold with manual maneuvers	1.5		1.5	3.0
SCS rotation		1069.0		1069.0
Free drift		7.5		7.5
Entry			0.3	0.3
Total time				**1080.0**

[redacted]

Table 88. SCS System Summary, Flight 221

Component	Weight Per Unit (Pounds)	Volume Per Unit (Cu In)	Number for Basic System	Number of Spares	Weight Per System (Pounds)	Volume Per System (Cu In)
Solenoid driver assembly	12.2	476	1	1	24.4	952
Control electronics assembly	16.1	495	1	-	16.1	495
Display electronics assembly	19.6	615	1	-	19.6	615
Gyro display coupler	15.7	430	1	-	15.7	430
Servo amplifier (TVC)	14.6	430	1	-	14.6	430
Gyro assembly	17.5	525	2	1	52.5	1575
Flight director attitude indicator	7.9	428	2	-	15.8	856
Gimbal position and fuel pressure indicator	3.7	150	1	-	3.7	150
Attitude set control panel	3.7	115	1	-	3.7	115
Translation control	4.3	70	1	-	4.3	70
Rotation control	6.2	94	2	-	12.4	188
Entry monitor system	5.0	112	1	0	5.0	112
Totals					187.8	5988

Modifications: Spares required

NOTE: Basic system reliability = 0.9501. Indicated spares raise reliability above 0.98.

[redacted]

[redacted]

Table 89. Service Module - RCS Requirements, Flight 221C

Mission Phase \ Requirement	Maximum Burn Time per Jet (Second)			Maximum Number Starts per Jet			Propellant Weight (Pounds)		
	Roll	Pitch	Yaw	Roll	Pitch	Yaw	Roll	Pitch	Yaw
Ascent	7.5	65.0	86.0	140	115	115	36.0	65.0	65.0
Orbit-keeping	0	98.0	98.0				0	145.0	145.0
In-orbit	1065.2	10.0	10.0	19	67	67	1578.3	15.0	15.0
Pre-entry	1.7	13.0	12.5	180	133	133	2.3	9.6	8.7
Totals	1074.4	186.0	206.5	339	315	315	1616.6	234.6	233.7

Table 90. Service Module - RCS Requirements, Flight 221D

Mission Phase \ Requirement	Maximum Burn Time per Jet (Second)			Maximum Number Starts per Jet			Propellant Weight (Pounds)		
	Roll	Pitch	Yaw	Roll	Pitch	Yaw	Roll	Pitch	Yaw
Ascent	7.5	65.0	86.0	140	115	115	36.0	65.0	65.0
Orbit-keeping	0	98.0	98.0				0	145.0	145.0
In-Orbit	1065.2	10.0	10.0	19	67	67	1578.3	15.0	15.0
Pre-entry	1.8	13.7	13.3	180	133	133	3.0	9.9	8.9
Totals	1074.5	208.2	207.3	339	315	315	1617.3	234.9	233.9

[redacted]

FLIGHT 518

Mission duration: 45 days. Orbit inclination: 83 degrees retrograde. Orbit altitude: 200 nautical miles. Crew size: 3 men.

Table 91. Control Requirements Summary, SCS, Flight 518

Experiment Number	Axis	Orientation Accuracy (Degrees)	Angular Velocity Limit (deg/sec)	Angular Acceleration Limit (deg/sec/sec)	Time Per Mission (Hours)	Operating Mode
0101-0102 Zero-G mode	All	NA	Minimum available	NA	3.2	SCS rate control (minimum impulse)
0101-0102 Rotation	All	NA	See profile	NA	8.0	SCS rotation
0101-0102 Linear acceleration	All	NA	Minimum	NA	1.6	SCS TVC
1405	2 axes Other	±10 DNA	NA	NA	789.2	Sun orient
0802A	All	±0.5	±0.5	0.5	60	SCS local vertical and manual
0802B	All	±0.5	±0.5	0.5	90	SCS local vertical and manual
0802C	All	±0.15	±0.5	No requirement	22.5	G&N prime with manual maneuvers
0802D	All	±0.5	±1.0	No requirement	0.5	G&N prime
0901	All	±0.1	±0.01	No requirement	101.5	G&N prime with manual maneuver

NA - Not applicable

Table 92. Rotational Maneuver Summary, Flight 518

Experiment Number	Axis	Slewing Rate (Degrees Per Second)	Maneuver Mode		Number Per Mission
			Manual	Automatic	
0101-0102	Pitch or yaw	Accelerate to 9 RPM in steps of 3 RPM each, and decelerate in similar manner	✓		19
0901	Roll and yaw	0.5*	✓		65
1405	Roll and pitch or yaw	0.5*	✓		342**
0802A	Roll and yaw	0.5*	✓		270
0802B	Roll and yaw	0.5*	✓		
0802C	Roll and yaw	0.5*	✓		
0802D	3 axis	0.5*			7

(*)NAA assumed
(**)Return to sun orient following each excursion (assumes A, B, C run sequentially when possible)

Table 93. Operating Mode - Time Summary, Flight 518

Mode of Operation	Time for Ascent (Hours)	Time on Orbit (Hours)	Time for Entry (Hours)	Total Time (Hours)
Ascent	0.2			0.2
SCS attitude hold with manual maneuvers	1.5		1.5	3.0
SCS TVC		1.6		1.6
SCS rotation		8.0		8.0
Sun orient		789.2		789.2
SCS local vertical with manual maneuvers		150.0		150.0
G&N prime with manual maneuvers		124.0		124.0
G&N prime		0.5		0.5
Free drift		0.0		0.0
SCS rate control (minimum impulse)		3.2		3.2
Entry			0.3	0.3
Total time				1080.0

Table 94. SCS System Summary, Flight 518

Component	Weight Per Unit (Pounds)	Volume Per Unit (Cu In)	Number for Basic System	Number of Spares	Weight Per System (Pounds)	Volume Per System (Cu In)
Solenoid driver assembly	12.2	476	1	1	24.4	952
Control electronics assembly	16.1	495	1	1	32.2	990
Display electronics assembly	19.6	615	1	-	19.6	615
Gyro display coupler	15.7	430	1	-	15.7	430
Servo amplifier (TVC)	14.6	430	1	-	14.6	430
Gyro assembly	17.5	525	2	-	35.0	1050
Flight director attitude indicator	7.9	428	2	-	15.8	856
Gimbal position and fuel pressure indicator	3.7	150	1	-	3.7	150
Attitude set control panel	3.7	115	1	-	3.7	115
Translation control	4.3	70	1	-	4.3	70
Rotation control	6.2	94	2	-	12.4	188
Entry monitor system	5.0	112	1	-	5.0	112
Horizon scanners and electronics	14.0	252	1	1	28.0	504
Totals					214.4	6462

Modifications: Add horizon scanners and associated electronics.
Add indicated spaces.

NOTE: Basic system reliability = 0.9567.
Indicated sparing raises reliability above 0.98

[redacted]

Table 95. Service Module - RCS Requirements, Flight 518C

Requirement / Mission Phase	Maximum Burn Time per Jet (Second)			Maximum Number Starts per Jet			Propellant Weight (Pounds)		
	Roll	Pitch	Yaw	Roll	Pitch	Yaw	Roll	Pitch	Yaw
Ascent	6.6	55	72.5	140	115	115	8.6	60.2	62.5
Orbit-keeping	0	45.5	45.5				0	67.5	67.5
In-Orbit	430	246.0	676.0	26,880	11,344	26,328	638.7	365.0	1002.5
Pre-entry	1.7	13.2	12.6	180	133	133	2.6	9.7	8.7
Totals	438.3	359.7	806.6	27,200	11,592	26,576	649.9	502.4	1141.2

Table 96. Service Module - RCS Requirements, Flight 518D

Requirement / Mission Phase	Maximum Burn Time per Jet (Second)			Maximum Number Starts per Jet			Propellant Weight (Pounds)		
	Roll	Pitch	Yaw	Roll	Pitch	Yaw	Roll	Pitch	Yaw
Ascent	6.8	56.5	74	140	115	115	9.6	60.9	63.7
Orbit-keeping	0	45.5	45.5				0	67.5	67.5
In-Orbit	288	314	674	22,235	10,960	23,138	595.4	392.9	998.6
Pre-entry	1.8	14	13.5	180	133	133	3.2	10	9
Totals	296.6	430	807	22,555	11,208	23,386	608.2	531.3	1138.8

[redacted]

FLIGHT 229

Mission duration: 45 days. Orbit inclination: 28.5 degrees. Orbit altitude: 200 nautical miles. Crew size: 3 men.

Table 97. Control Requirements Summary, SCS, Flight 229

Experiment Number	Axis	Orientation Accuracy (Degrees)	Angular Velocity Limit (deg/sec)	Angular Acceleration Limit (deg/sec/sec)	Time Per Mission (Hours)	Operating Mode
0101-0102 Zero-G	All	NA	Minimum	NA	3.2	SCS rate control (minimum impulse)
0101-0102 Rotation	All	NA	See profile	NA	8.0	SCS rotation
0101-0102 Linear acceleration	All	NA	Minimum	NA	1.6	SCS TVC
1504	All	±5	±0.2	NA	28.5	SCS attitude hold with manual maneuver
1305	All	±5	±0.2	NA	6	SCS attitude hold with manual maneuver
1302	All	±5	±0.2	NA	2	SCS attitude hold with manual maneuver
1306	All	±5	±0.2	NA	0.5	SCS attitude hold with manual maneuver
1306	All	±0.3	±.02	NA	6.0	G&N prime
1301	All	±5	±0.2	NA	3.3	SCS attitude hold with manual maneuver
0703	All	±0.5	No requirement	No requirement	7.5	SCS attitude hold
0601	All	NA	±0.2*	NA	11.2	SCS rate control (minimum impulse)
0602A	All	NA	±0.2*	10^{-4}	22.5	SCS rate control (minimum impulse)

(*)Actual requirement stated as 10^{-4}g equivalent

Table 97. Control Requirements Summary, SCS, Flight 229 (Cont)

Experiment Number	Axis	Orientation Accuracy (Degrees)	Angular Velocity Limit (deg/sec)	Angular Acceleration Limit (deg/sec/sec)	Time Per Mission (Hours)	Operating Mode
0602B	All	NA	±0.2**	10^{-4}	6	SCS rate control (minimum impulse)
0603	All	NA	±0.2*	10^{-4}	2.5	SCS rate control (minimum impulse)
0604	All	NA	±0.2**	Minimal**	4	SCS rate control (minimum impulse)
0605	All	±0.5***	±0.2	Minimal**	11.2	SCS attitude hold (minimum impulse)
0606			Assume will not schedule			

(*)Actual requirement stated as 10^{-4}g equivalent
(**)NAA assumed
(***)Solar orientation
NA - Not applicable

Table 98. Rotational Maneuver Summary, Flight 229

Experiment Number	Axis	Slewing Rate (Degrees Per Second)	Maneuver Mode		Number Per Mission
			Manual	Automatic	
0101-0102	Pitch or yaw	Accelerate to 9 RPM in steps of 3 RPM each, and decelerate in similar manner	✓		19
1504	Pitch and yaw	0.5*	✓		36
1305	Pitch and yaw	0.5*	✓		17
1302	Pitch and yaw	0.5*	✓		8
1306	3 axis	0.5*		✓	13
1301	Pitch and yaw	0.5*	✓		2
0703	Roll and pitch or yaw	0.5*	✓		15
0605	Roll and pitch or yaw	0.5*	✓		22

*NAA assumed

Table 99. Operating Mode - Time Summary, Flight 229

Mode of Operation	Time for Ascent (Hours)	Time on Orbit (Hours)	Time for Entry (Hours)	Total Time (Hours)
Ascent	0.2			0.2
SCS attitude hold with manual maneuvers	1.5	40.3	1.5	43.3
SCS rate control (minimum impulse)		68.1		68.1
SCS rotation		8.0		8.0
SCS TVC		1.6		1.6
G&N prime		6.0		6.0
Free drift		952.5		952.5
Entry			0.3	0.3
Total Time				1080.0

Table 100. SCS System Summary, Flight 229

Component	Weight Per Unit (Pounds)	Volume Per Unit (Cu In)	Number for Basic System	Number of Spares	Weight Per System (Pounds)	Volume Per System (Cu In)
Solenoid driver assembly	12.2	476	1	-	12.2	476
Control electronics assembly	16.1	495	1	-	16.1	495
Display electronics assembly	19.6	615	1	-	19.6	615
Gyro display coupler	15.7	430	1	-	15.7	430
Servo amplifier (TVC)	14.6	430	1	-	14.6	430
Gyro assembly	17.5	525	2	-	35.0	1050
Flight director attitude indicator	7.9	428	2	-	15.8	856
Gimbal position fuel pressure indicator	3.7	150	1	-	3.7	150
Attitude set control panel	3.7	115	1	-	3.7	115
Translation control	4.3	70	1	-	4.3	70
Rotation control	6.2	94	2	-	6.2	94
Entry monitor system	5.0	112	1	-	5.0	112
Totals					158.1	4987

Modifications: Provide capability for obtaining attitude information from both gyro sets (additional switching).

NOTE: Reliability of basic system (no mod.) = 0.9780.

Reliability with modification = 0.9954.

Table 101. Service Module - RCS Requirements, Flight 229C

Mission Phase \ Requirement	Maximum Burn Time per Jet (Second)			Maximum Number Starts per Jet			Propellant Weight (Pounds)		
	Roll	Pitch	Yaw	Roll	Pitch	Yaw	Roll	Pitch	Yaw
Ascent	4.8	40	52.5	140	115	115	7	43.6	45
Orbit-keeping	0	23.6	23.6				0	35.0	35.0
In-Orbit	24.2	670	670	1659	3716	3713	35.9	988.5	988.5
Pre-entry	1.7	13.2	12.8	180	133	133	2.5	9.7	8.8
Totals	30.7	746.8	758.9	1979	3964	3961	45.4	1076.8	1077.3

Table 102. Service Module - RCS Requirements, Flight 229D

Mission Phase \ Requirement	Maximum Burn Time per Jet (Second)			Maximum Number Starts per Jet			Propellant Weight (Pounds)		
	Roll	Pitch	Yaw	Roll	Pitch	Yaw	Roll	Pitch	Yaw
Ascent	4.7	39.5	51.8	140	115	115	5.9	43	44.7
Orbit-keeping	0	23.6	23.6				0	35	35
In-Orbit	25.7	705	705	1558	4169	4166	38.7	998	998
Pre-entry	1.8	13.7	13.3	180	133	133	3	9.9	8.9
Totals	32.2	781.8	793.7	1878	6295	6292	47.6	1085.9	1086.6

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Flight 230

Mission duration: 45 days. Orbit inclination: 28.5 degrees. Orbit altitude: 200 nautical miles. Crew size: 3 men.

Table 103. Control Requirements Summary, SCS, Flight 230

Experiment Number	Axis	Orientation Accuracy (Degrees)	Angular Velocity Limit (deg/sec)	Angular Acceleration Limit (deg/sec/sec)	Time Per Mission (Hours)	Operating Mode
0101-0102 Zero-g	All	NA	Minimum	NA	3.2	SCS rate control (minimum impulse)
0101-0102 Rotation	All	NA	See profile	NA	8.0	SCS rotation
0101-0102 Linear acceleration	All	NA	Minimum	NA	1.6	SCS TVC
1305	All	±5	±0.2	NA	6	SCS attitude hold with manual maneuvers
1302	All	±5	±0.2	NA	2	SCS attitude hold with manual maneuvers
1306	All	±5	±0.2	NA	0.5	SCS attitude with manual maneuvers
1306	All	±0.3	±.02	NA	6.0	G&N prime
1301	All	±5	±0.2*	NA	3.3	SCS attitude hold with manual maneuvers
0703	All	±0.5	No requirement	No requirement	7.5	SCS attitude hold
0601	All	NA	±0.2*	NA	11.2	SCS rate control (minimum impulse)
0602A	All	NA	±0.2**	10^{-4}	22.5	SCS rate control (minimum impulse)
0602	All	NA	±0.2**	10^{-4}	6	SCS rate control (minimum impulse)

(*)Actual requirement stated as 10^{-4}g equivalent
(**)NAA assumed, no requirement specified
(***)Solar orientation
NA - Not applicable

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[redacted]

Table 103. Control Requirements Summary, SCS, Flight 230 (Cont)

Experiment Number	Axis	Orientation Accuracy (Degrees)	Angular Velocity Limit (deg/sec)	Angular Acceleration Limit (deg/sec/sec)	Time Per Mission (Hours)	Operating Mode
0603	All	NA	±0.2*	10^{-4}	2.5	SCS rate control (minimum impulse)
0604	All	NA	±0.2**	Minimal**	4	SCS rate control (minimum impulse)
0605	All	±0.5***	±0.2	Minimal**	11.2	SCS attitude hold (mission impulse)
0606			Assume will not schedule			
1506 and 1504	All	±5	±0.2	No requirement	4.8	SCS attitude hold with manual maneuvers

(*)Actual requirement stated as 10^{-4}g equivalent
(**)NAA assumed, no requirement specified
(***)Solar orientation
NA - Not applicable

Table 104. Rotational Maneuver Summary, Flight 230

Experiment Number	Axis	Slewing Rate (Degrees Per Second)	Maneuver Mode		Number Per Mission
			Manual	Automatic	
0101-0102	Pitch or yaw	Accelerate to 9 RPM in steps of 3 RPM each, and decelerate in similar manner	√		19
1504 and 1506	Pitch and yaw	0.5	√		5
1305	Pitch and yaw	0.5	√		17
1302	Pitch and yaw	0.5	√		8
1306	3 axis	0.5		√	13
1301	Pitch and yaw	0.5	√		2
0703	Roll and pitch or yaw	0.5	√		15
0605	Roll and pitch or yaw	0.5	√		22

*NAA assumed

[redacted]

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Table 105. Operating Mode - Time Summary, Flight 230

Mode of Operation	Time for Ascent (Hours)	Time on Orbit (Hours)	Time for Entry (Hours)	Total Time (Hours)
Ascent	0.2			0.2
SCS attitude hold with manual maneuvers	1.5	16.6	1.5	19.6
SCS rate control (minimum impulse)		68.1		68.1
SCS rotation		8.0		8.0
SCS TVC		1.6		1.6
G&N prime		6.0		6.0
Free drift		976.2		976.2
Entry			0.3	0.3
Total Time				1080.0

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Table 106. SCS System Summary, Flight 230

Component	Weight Per Unit (Pounds)	Volume Per Unit (Cu In)	Number for Basic System	Number of Spares	Weight Per System (Pounds)	Volume Per System (Cu In)
Solenoid driver assembly	12.2	476	1	-	12.2	476
Control electronics assembly	16.1	495	1	-	16.1	495
Display electronics assembly	19.6	615	1	-	19.6	615
Gyro display coupler	15.7	430	1	-	15.7	430
Servo amplifier (TVC)	14.6	430	1	-	14.6	430
Gyro assembly	17.5	525	2	-	35.0	1050
Flight director attitude indicator	7.9	428	2	-	15.8	856
Gimbal position and fuel pressure indicator	3.7	150	1	-	3.7	150
Attitude set control panel	3.7	115	1	-	3.7	115
Translation control	4.3	70	1	-	4.3	70
Rotation control	6.2	94	2	-	6.2	94
Entry monitor system	5.0	112	1	-	5.0	112
Totals					158.1	4987

Modifications: None

NOTE: Reliability of basic system = 0.9831

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Table 107. Service Module - RCS Requirements, Flight 230C

Mission Phase \ Requirements	Maximum Burn Time per Jet (Second)			Maximum Number Starts per Jet			Propellant Weight (Pounds)		
	Roll	Pitch	Yaw	Roll	Pitch	Yaw	Roll	Pitch	Yaw
Ascent	4.8	40	52.5	140	115	115	6.9	43.4	44.7
Orbit-keeping	0	23.6	23.6				0	35	35
In-Orbit	21.3	678	678	1649	2936	2933	31.5	1011	1011
Pre-entry	1.7	13	12.5	180	133	133	2.3	9.6	8.7
Totals	27.8	754.6	766.6	1969	3184	3181	40.7	1099.0	1099.4

Table 108. Service Module - RCS Requirements, Flight 230D

Mission Phase \ Requirement	Maximum Burn Time per Jet (Second)			Maximum Number Starts per Jet			Propellant Weight (Pounds)		
	Roll	Pitch	Yaw	Roll	Pitch	Yaw	Roll	Pitch	Yaw
Ascent	4.7	39.5	51.8	140	115	115	5.9	43	44.7
Orbit-keeping	0	23.6	23.6				0	35	35
In-Orbit	22.8	675	675	1547	3413	3410	33.7	1000	1000
Pre-entry	1.7	13.5	13	180	133	133	2.9	9.9	8.8
Totals	29.2	751.6	763.4	1867	3661	3658	42.5	1087.9	1088.5

APPENDIX B

ADDITIONAL REACTANTS FOR SM FUEL CELLS

Configuration 1 flights are based on a 14-day mission duration. Providing for this duration without additional batteries depends on the total electrical energy requirements, average load throughout the mission, efficiency of the fuel cells, and the amount of reactants available for fuel cell operations. The reactants in the CSM Configuration 1 alone are not sufficient to provide the total energy requirements for the 14-day flights. The use of primary batteries in the external device to extend the mission to 14 days is certainly feasible, but the total weight allotment of these batteries may be as high as 1800 pounds. Therefore, other methods were also considered that would produce much more energy per pound of added weight. These additional methods include installation of additional cryogenic tanks in Sector IV, and a fluid interconnection between the service module and additional tanks in the external device.

The Apollo cryogenic storage system contains hydrogen and oxygen for the fuel cells and additional oxygen for the environmental control system. In the AES configurations, the cabin atmosphere in the external device is interconnected with the command module atmosphere, permitting ECS makeup oxygen to be supplied from either the external device or the service module. This feature makes it feasible to divert the ECS oxygen in the service module to fuel cell use, and supply the ECS from the external device. Thus, for the AES configuration, electrical energy from the service module is limited by hydrogen availability rather than by hydrogen and oxygen quantities.

In the Apollo X study it was established that one additional hydrogen tank and one additional oxygen tank could be installed in Sector IV of the service module, although such installation was crowded and required that one tank must extend through the lower bulkhead. The addition of one hydrogen tank only is significantly easier from a physical packaging standpoint, but even this requires relocation of the tanks, plumbing, and shelves in Section IV, and is, therefore, more of a major modification than the ground rules of this study allow. The total electrical energy available would be increased almost 50 percent by using all service module oxygen for fuel cells and adding one additional Block II hydrogen tank.

Another method of obtaining additional hydrogen for the fuel cells is by installation of additional hydrogen tanks in the external device. This method requires that gaseous (not necessarily cryogenic) hydrogen connections be made from the external device to the service module. The rest of Appendix B is concerned with the discussion of this type of utility line. This utility line is characterized as follows:

1. It must be controllable by the flight crew.
2. It cannot interfere with heat and pressure balance in either CGS.
3. It cannot interfere with heat and pressure balance in either of the fuel cell powerplants.
4. Reliability must be high.
5. It must include quick-connect/quick-disconnect which the astronaut can manipulate easily with one hand.
6. It cannot pass through the oxygen atmosphere of the command module.

After the docking position has been obtained, an astronaut climbs out of the command module and quick-connects the flexible line at the location shown (Figure 99). Only one hand is required for this operation. Solenoid

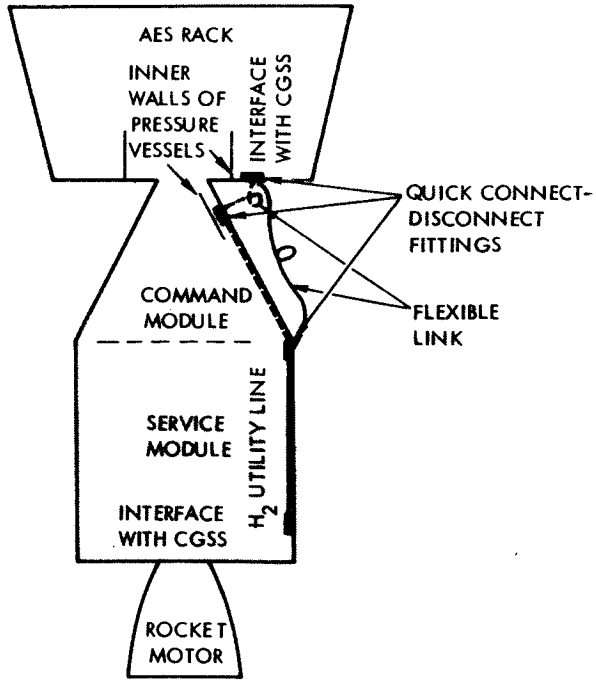

Figure 99. Location of Hydrogen Utility Line Between Cryogenic Gas Storage Systems

valves at each end of the hydrogen utility line isolate it from cryogenic storage, until the line is required for fluid transport.

The hydrogen utility line comprises four links: (1) the service module (cryogenic gas storage) line; (2) the transport line, routed along the service module and possibly along the command module; (3) the flexible link between the command or service module and the external device; and (4) the line in the external device.

Two basic routes were considered for the transport line along the service module and the command module: (1) outside the spacecraft, and (2) between the inner and outer walls, both shown in Figure 99.

If the transport line is located outside the spacecraft, installation by a single astronaut in a "hard" suit seems feasible. Direct solar heat on the line may possibly raise the fluid temperature and pressure above the acceptable limits of 130 F and 250 psi, unless the line is protected by suitable insulation.

If the line is located between the inner and outer walls of the command module, as shown in Figure 99, objections to the first method are removed; but more extensive modification of the command module is required in addition to some method of cutting the line upon command-module service-module separation. The use of a longer, flexible line from the external device to the service module eliminates any possible compromise of command module integrity.

Quick-connect/quick-disconnect fittings compatible with hydrogen are available. Although no fittings are space-rated, those now in use for aircraft applications may be expected to qualify after modification, development necessary for space qualification need only be moderate. Typical temperature and pressure characteristics of highly reliable off-the-shelf units are as follows:

Static operating pressure: 3000 psi

Impulse requirements: 4500 psi (MIL C 25427)

Temperature range: -65 to +275 F

Flexible line-coupling between the service module and the AES external device cryogenic gas storage systems (Figure 99) is most conveniently located outside the spacecraft in the space environment. Although a development program is required to qualify this equipment fully for its intended application, the flexible link, as presently conceived, consists of 1/4-inch stainless tube, the standard flared ends equipped with adapters

which are matched with quick-connect/quick-disconnect fittings. This flexible link is preformed and prewrapped with reflective foil, so that in making the connection the astronaut has only to align the part and connect first one then the other end by a simple forward thrust requiring only the use of one hand. A wrapping which may be suitable for the application is 0.000010-inch gold plate over either 0.001-inch molybdenum foil or 0.005-inch dimpled tantalum foil — each to be prepared according to plating processes developed at the Atomics International Division of NAA.

Cryogenic gas storage system plumbing configurations would be essentially the same in both the Apollo service module and any CGS system in the AES external device. Plumbing configurations within 400 hour-rated and 1000 hour-rated fuel cell modules are also essentially the same.

The CGS plumbing configuration, typical of both service module and external device, is shown in schematic form in Figure 100. The numbers 1, 2, 3, and 4 indicate points where line connections seem feasible. Since fluid flow in the hydrogen utility line may be either from the service module to the external device if fuel cells are installed in the external device, or vice-versa, depending on the need for fluid, connections 1, 2, 3, and 4 in the service module and the external device have equal significance. These connections were examined by using the matrix of Table 109 to verify that all requirements were met.

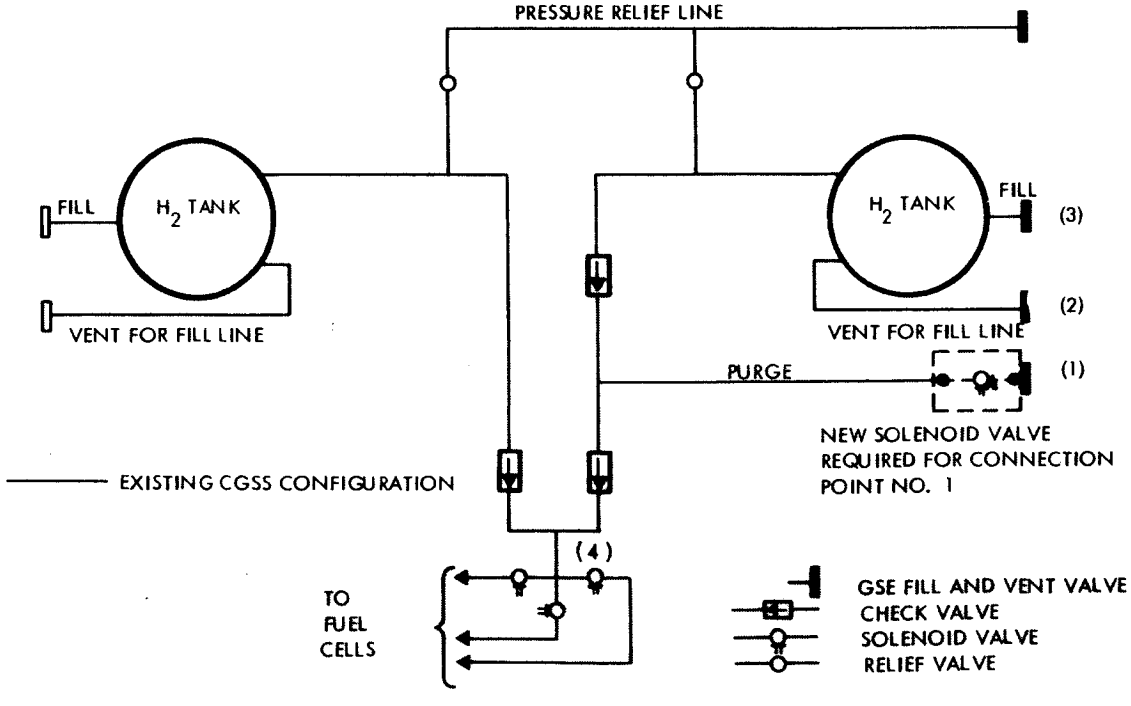

Figure 100. CGSS Configuration Typical of Installations in Both Apollo Service Module and AES Rack, With Possible Connections for Hydrogen Utility Line

Table 109. Matrix of Possible Hydrogen Utility Line Connections and System Requirements

Connection Number		Meet Requirements Number						
In SM or AES, Respectively	In AES or SM, Respectively	1	2	3	4	5	6	All
1	1	X	X	X	X	X	X	X
1	2	X		X	X	X	X	
1	3	X		X	X	X	X	
1	4	X	X	X	X	X	X	X
2	1	X		X	X	X	X	
2	2	X		X	X	X	X	
2	3	X		X	X	X	X	
2	4	X		X	X	X	X	
3	1	X		X	X	X	X	
3	2	X		X	X	X	X	
3	3	X		X	X	X	X	
3	4	X		X	X	X	X	
4	1	X	X	X	X	X	X	X
4	2	X		X	X	X	X	
4	3	X		X	X	X	X	
4	4	X	X	X	X	X	X	X

Connections 1 and 4 and all combinations thereof meet requirements as listed. Hybrid connections of the 1-to-4 add complexity with no advantage over the simple connections of 1-to-1 or 4-to-4. Therefore, the simple type of connection is preferred. Connection 1 is the most reliable, providing that storage tanks to which they are connected are operating normally. If this is not the case, the utility line is rendered inoperable. Since all storage tanks are tapped, connections to points 4 are most reliable and the fluid flow is most stable — as long as the common manifold to which the utility line is joined operates without malfunction. If the common manifold leaks, all storage tanks in both service module and external device can become depleted. This is an example of single-mode failure.

For all configurations, the hydrogen utility line should have solenoid valves at each end; thus, in the event of malfunction, the line could not deplete the contents of any hydrogen storage tank.

Other than connections 1 and 4, many hybrid configurations in corresponding elements of both types (1 and 4) may be considered; for example, the valves. The utility line is connected between these valves (Figure 101). Another hybrid connection may be a modification of Type 1, using more than one line connection (Figure 102). Further modifications of the two basic configurations are possible. Each configuration has some advantageous and some disadvantageous features.

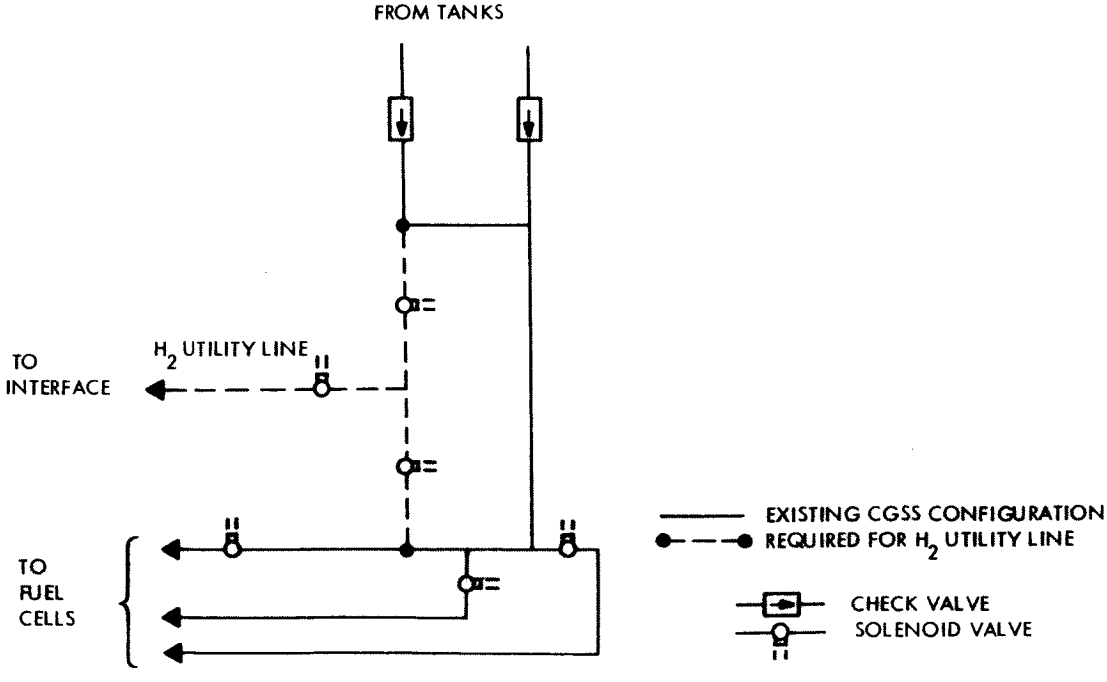

Figure 101. Example of Modified Common Manifold Configuration - Connection to Hydrogen Utility Line

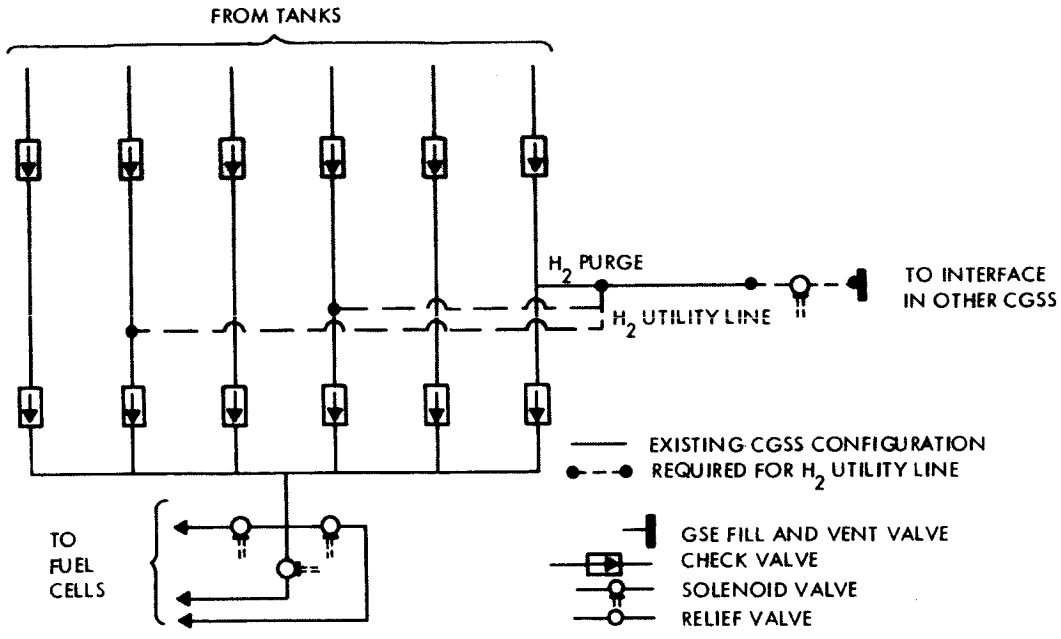

Figure 102. Example of Modified Hydrogen Purge Configuration

Since one line of the simple 1-to-1 type of configuration (e.g., the purge line of Figure 100) is existing, CGS hardware requiring only the installation of a solenoid valve for isolation of the utility line (1-to-1 configuration) is preferred. With two hydrogen tanks comprising the storage system, another line similar to the modified purge line is installed (Figure 103). For one- and two-tank storage systems, this simple 1-to-1 configuration is superior to any of the configurations and modifications previously considered for the following reasons: a minimum modification and change effort is required, and all necessary changes (T-joints, straight line coupling, installation of solenoid valves) involve processes already developed on the existing CGS.

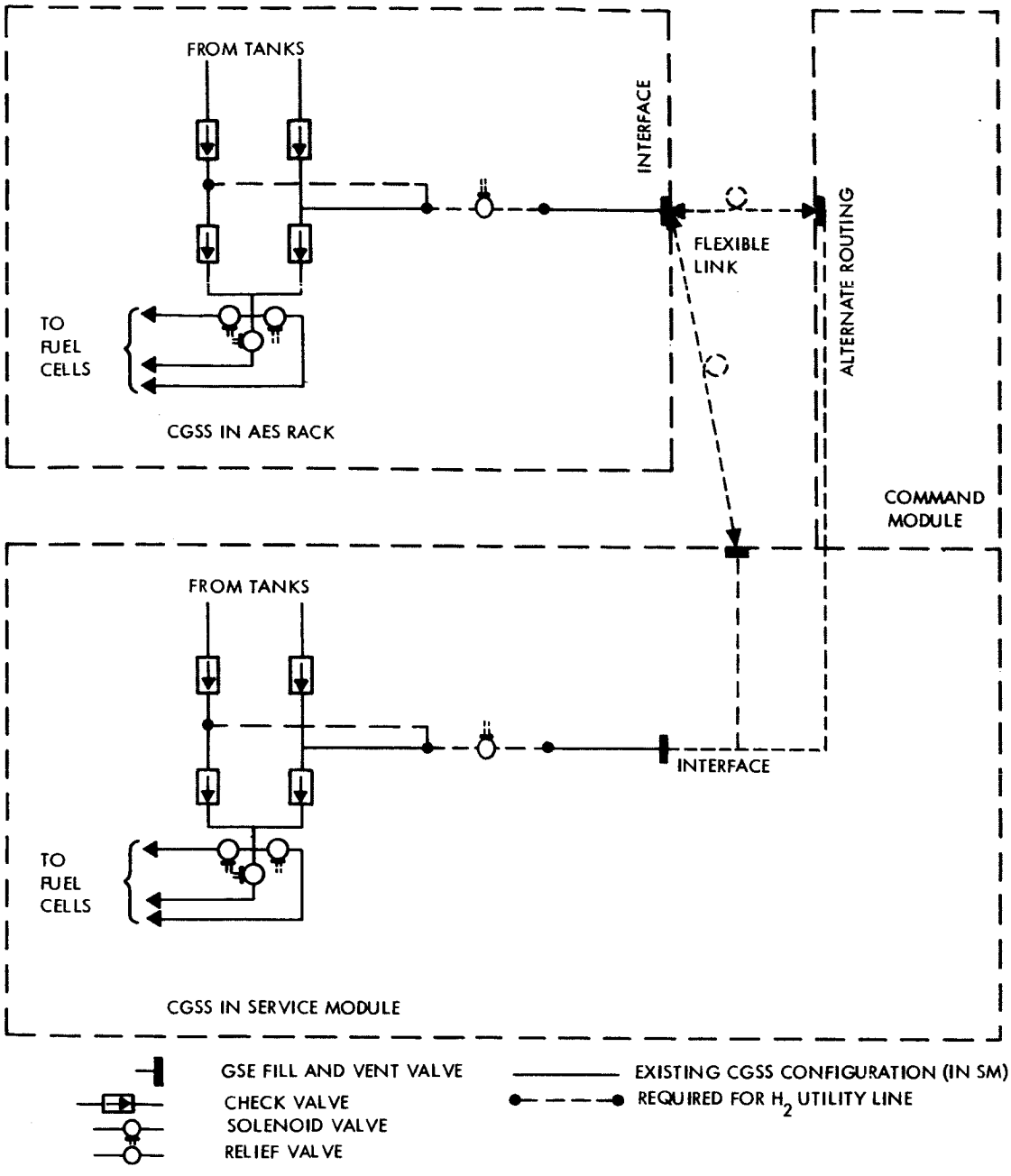

Figure 103. Hydrogen Utility Line Configuration - Two Hydrogen Tanks in Each CGSS